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NASA Conference Publication 2144

ORIGINAL
COMPLETED

Large Space Systems/Low-Thrust Propulsion Technology

An Industry-Government
Information Exchange held
at Lewis Research Center
Cleveland, Ohio, May 20-21, 1980

NASA Conference Publication 2144

Large Space Systems/Low-Thrust Propulsion Technology

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Information Exchange
organized by and held at
NASA Lewis Research Center
Cleveland, Ohio, May 20-21, 1980

National Aeronautics
and Space Administration

**Scientific and Technical
Information Branch**

1980

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PREFACE

Many Shuttle-era missions consist of spacecraft that are deployed or assembled to dimensions that are many times larger than the dimensions of the Orbiter's cargo bay. In consideration of this class of missions, NASA has embarked on a Large Space System Technology (LSST) Program, a multicenter program lead by NASA Langley, whose purpose is to identify, address, and solve problems to allow large spacecraft systems to become the basis for future missions.

The purpose of this meeting was to address the potentially critical interactions that occur between propulsion, structures and materials, and controls for large spacecraft; to define the technology impacts within these fields; and to determine the net effect on large space systems and the resulting missions. Presentations were made in three topical areas: Systems/Mission Analysis, LSS Static and Dynamic Characterization, and Propulsion System Characterization. The intent of this was to provide an interdiscliplinary exchange of information between propulsion, structures and materials, and controls, with emphasis on large spacecraft and missions. A summary of the issues raised and information supplied in the presentations was accomplished in an open discussion period at the end of the meeting.

The meeting was attended by 85 people representing NASA Headquarters, 5 NASA Centers, 5 DOD Organizations, 14 Aerospace Companies, and 2 National Laboratories. The 22 presentations made and minutes of the open discussion are compiled in this publication.

Richard F. Carlisle
NASA Headquarters

Meeting Chairman

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INTRODUCTION: THE CHALLENGE OF OPTIMUM INTEGRATION OF PROPULSION SYSTEMS AND LARGE SPACE STRUCTURES

Richard F. Carlisle
NASA Headquarters

The integration of propulsion systems and large space structures systems will result in an optimum spacecraft system design that will provide an improved facility and resources to an onboard payload designed to meet mission requirements. Characteristics of each system will be discussed and technology challenges will be identified.

Introduction

The Spacecraft Systems Office's goal, Figure 1, is to define and implement new technology tasks that will provide cost effective operational spacecraft for the 1990's that meet new challenging mission performance requirements at an affordable reduced cost. The office addresses three classes of spacecraft: large space systems at Low Earth Orbit (LEO); advanced spacecraft at Geostationary Earth Orbit (GEO); and advanced planetary spacecraft. This paper discusses the integration of propulsion system and structure systems primarily at LEO and GEO and the transfer task from LEO to GEO.

The purpose of this meeting is to provide a technology exchange of the state-of-the-art and system characteristics of the two systems in question, that is propulsion and structures. It is envisioned that when we each have a better understanding of the design characteristic constraints and sensitivities of each other's technology, we will be able to offer ideas and suggestions of trade-offs that will benefit in an optimized integrated design, Figure 2.

This meeting will be successful if we can surface technology questions and/or concerns that result in challenges and action items for future consideration. Your attendence here today represents the experts in the industry in these two disciplines.

I charge each of you to be attentive and give it your best for two days and make this technology exchange a practical contribution that will result in better, lower cost spacecraft to meet the requirements of future challenging missions at affordable cost.

Integrated Propulsion and Structures Sub-System Functional Matrix (Figure 3)

The most significant external disturbance of a large space system in low earth orbit is aerodynamic drag that must be compensated for by some type of mass expulsion actuator. Aerodynamic drag predominates at altitudes below approximately 140-160 miles depending on the size and spacecraft configuration. The Shuttle has difficulty in carrying large spacecraft into high orbits. If it is desired to operate at say 200-240 miles a popular technique is to deploy the structure at a more convenient lower orbit and provide enough propulsion on board the spacecraft so that the spacecraft engines can put the spacecraft into a higher orbit.

The above scenario says if a spacecraft is of a given configuration and size it must have propulsion on board. This propulsion is required to provide multi burn, low thrust performance over many starts and stops for a long operational life. A major question then is, if this propulsion is on board as part of the spacecraft design what other requirements should be imposed on this system? If the spacecraft can provide for its own orbit maintenance and/or maneuvers, it can eliminate the need of the support of a costly transportation vehicle.

Figure 3 shows a functional matrix of possible propulsion system characteristics for a spacecraft for deployable and assembled spacecraft structures. The matrix shows that either electric propulsion or low thrust chemical propulsion systems could provide the propulsion required. The figure shows the trade-off considerations of a single propulsion engine or multi-engines. The figure illustrates that a single point engine is bounded by some upper limit of thrust for assembled spacecraft. The matrix also shows several additional functions that can be provided to the spacecraft if a propulsion system is an integral part of the spacecraft. For example, one may not include a propulsion system to a spacecraft design for momentum dump, however, if there is a propulsion system on board for stationkeeping or orbital maintenance it may well be used also for momentum dump. A careful review of all of the functions that can be provided for a spacecraft by an integral propulsion system may result in the inclusion of the propulsion for several functions even if no single function were mandatory.

The next figure (Figure 4) shows propulsion interface issues for each combination of engines discussed in the previous chart (function matrix Figure 3). A single engine has a single loading point into the structure that requires load carrying members into the structure from a hard point mechanical interface. Low thrust engines may excite structural dynamics that result in negative forces at the engine. This interaction represents an engine design constraint derived from the structural dynamics. In turn the propulsion dynamics must be compatible with structural dynamics or the engine may excite structural transients during engine starting and stopping.

Multiple engines introduce additional interface issue specifically relative to the sensing tolerance of the multiple engine dynamics. If engine starts are out of sync unpredicted structural response between engines could occur.

The next figure (Figure 5) illustrates advantages of each alternate propulsion configuration.

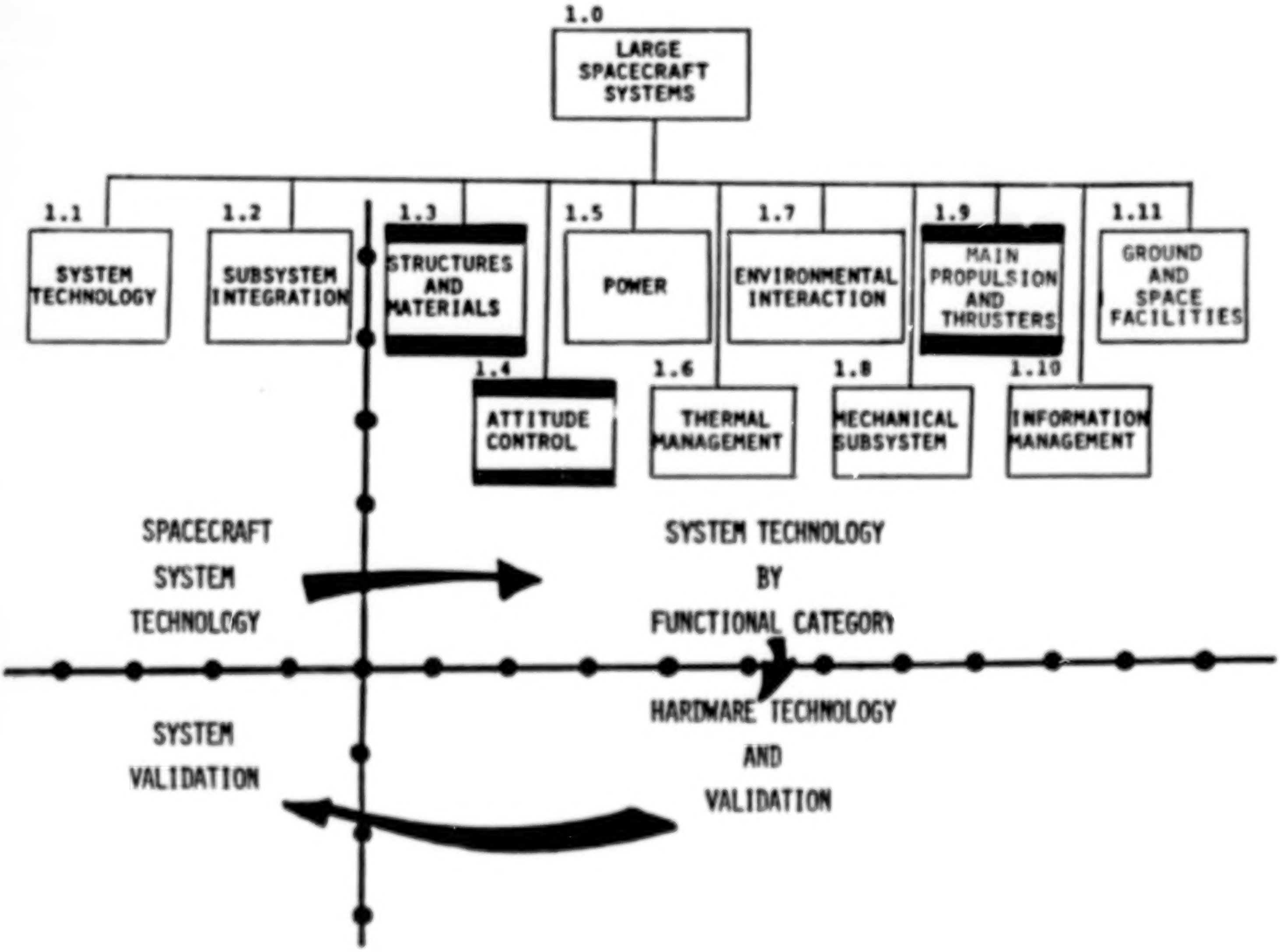
FIGURE 1 SPACECRAFT SYSTEMS

GOAL

- DEVELOP SUBSYSTEM TECHNOLOGIES FOR OPERATIONAL SPACECRAFT AND SPACE OPERATIONS FOR THE 1990'S

  - INCREASE CAPABILITIES
  - DECREASE COSTS

FIGURE 1 (Concluded) OAST LARGE SPACE SYSTEMS PROGRAM ELEMENTS AND RELATIONSHIP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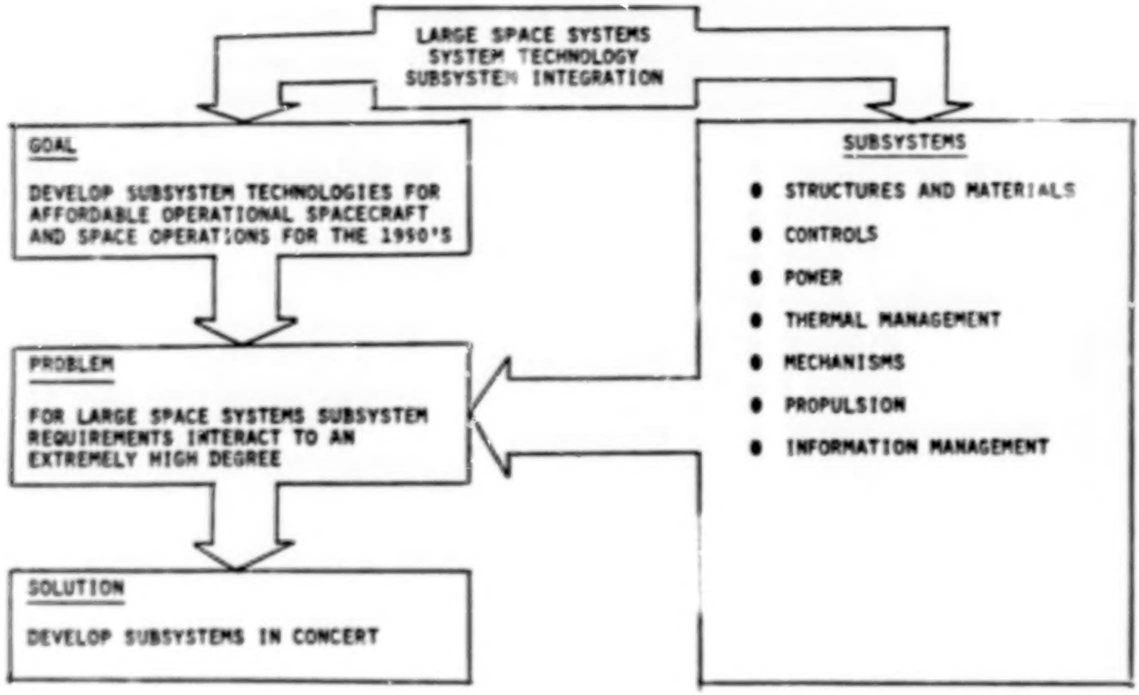
LARGE SPACECRAFT SYSTEMS
1.1
SYSTEM TECHNOLOGY
1.2
SUBSYSTEM INTEGRATION
1.3
STRUCTURES AND MATERIALS
1.4
ATTITUDE CONTROL
1.5
POWER
1.6
THERMAL MANAGEMENT
1.7
ENVIRONMENTAL INTERACTION
1.8
MECHANICAL SUBSYSTEM
1.9
MAIN PROPULSION AND THRUSTERS
1.10
INFORMATION MANAGEMENT
1.11
GROUND AND SPACE FACILITIES
SPACECRAFT SYSTEM TECHNOLOGY
SYSTEM TECHNOLOGY BY FUNCTIONAL CATEGORY
SYSTEM VALIDATION
HARDWARE TECHNOLOGY AND VALIDATION

FIGURE 2 SPACECRAFT SYSTEMS OFFICE/LARGE SPACE SYSTEMS



LARGE SPACE SYSTEMS
SYSTEM TECHNOLOGY
SUBSYSTEM INTEGRATION

GOAL

DEVELOP SUBSYSTEM TECHNOLOGIES FOR AFFORDABLE OPERATIONAL SPACECRAFT AND SPACE OPERATIONS FOR THE 1990'S

PROBLEM

FOR LARGE SPACE SYSTEMS SUBSYSTEM REQUIREMENTS INTERACT TO AN EXTREMELY HIGH DEGREE

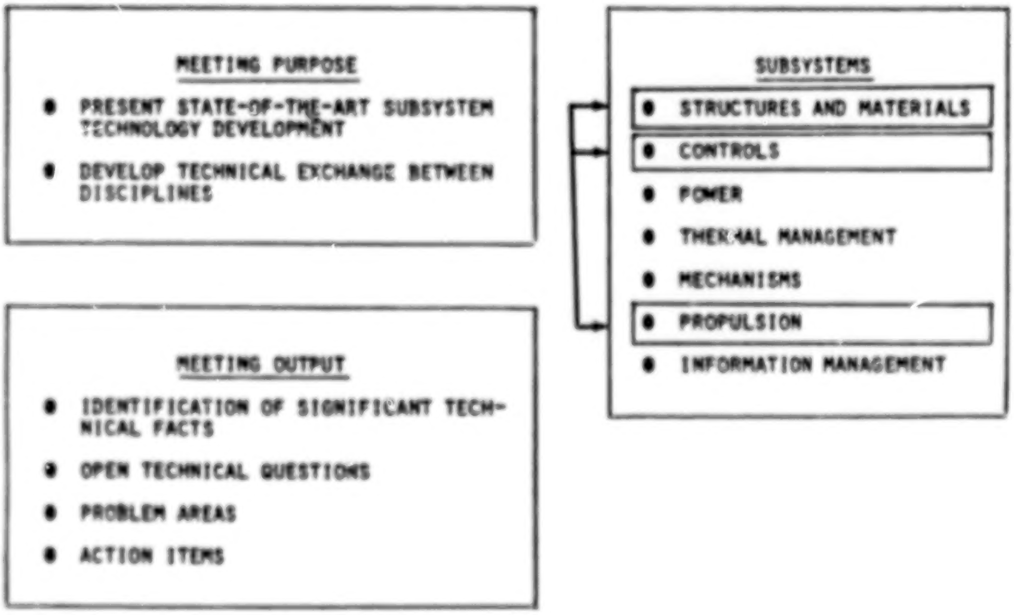
SOLUTION

DEVELOP SUBSYSTEMS IN CONCERT

SUBSYSTEMS

- STRUCTURES AND MATERIALS
- CONTROLS
- POWER
- THERMAL MANAGEMENT
- MECHANISMS
- PROPULSION
- INFORMATION MANAGEMENT

FIGURE 2 (Concluded) SPACECRAFT SYSTEMS OFFICE/LARGE SPACE SYSTEMS



MEETING PURPOSE

- PRESENT STATE-OF-THE-ART SUBSYSTEM TECHNOLOGY DEVELOPMENT
- DEVELOP TECHNICAL EXCHANGE BETWEEN DISCIPLINES

MEETING OUTPUT

- IDENTIFICATION OF SIGNIFICANT TECHNICAL FACTS
- OPEN TECHNICAL QUESTIONS
- PROBLEM AREAS
- ACTION ITEMS

SUBSYSTEMS

- STRUCTURES AND MATERIALS
- CONTROLS
- POWER
- THERMAL MANAGEMENT
- MECHANISMS
- PROPULSION
- INFORMATION MANAGEMENT

FIGURE 3 INTEGRATED PROPULSION AND STRUCTURES SUBSYSTEM FUNCTIONAL MATRIX*

STRUCTURES SUBSYSTEM
(INCREASING SIZE ⟶)

			DEPLOYABLE	ASSEMBLED
MAIN PROPULSION AND THRUSTER SUBSYSTEM (INCREASING THRUST LEVEL ↓)	ELECTRIC PROPULSION	SINGLE POINT	• ORBIT TRANSFER LEO TO HEO, AND LEO TO GEO IN PACKAGED OR DEPLOYED STATE • STATIONKEEPING AT LEO HEO, AND GEO	SAME ASSEMBLED ←SIZE/THRUST LIMIT
	LOW THRUST CHEMICAL	MULTI-POINT	• ORBIT TRANSFER LEO TO HEO AND LEO TO GEO IN DEPLOYED STATE	SAME IN ASSEMBLED STATE
			• STATIONKEEPING • ATTITUDE CONTROL, STABILITY AND POINTING • FIGURE CONTROL • MOMENTUM DUMP • END OF LIFE DISPOSAL	SAME
	IUS SUSS A/D	SINGLE POINT	• ORBIT TRANSFER LEO TO HEO AND LEO TO GEO IN PACKAGED STATE	

* TELEOPERATOR RENDEZVOUS FUNCTIONS OMITTED

FIGURE 4 INTEGRATED PROPULSION AND STRUCTURES SUBSYSTEM INTERFACE ISSUE MATRIX

STRICTURES SUBSYSTEM
(INCREASING SIZE ⟶)

			DEPLOYABLE	ASSEMBLED
MAIN PROPULSION AND THRUSTER SUBSYSTEMS (INCREASING THRUST LEVEL ↓)	ELECTRIC PROPULSION	SINGLE POINT	• SINGLE LOAD PATH INTO DEPLOYED STRUCTURE • NEGATIVE ACCELERATION INTO PROPULSION SYSTEMS THROUGH LOW FREQUENCY STRUCTURAL VIBRATION • EXCITATION OF LOW FREQUENCY STRUCTURES	SAME ASSEMBLED ←SIZE/THRUST LIMIT
	LOW THRUST CHEMICAL	MULTI-POINT	• INTEGRATION OF PROPULSION SUBSYSTEM ELEMENTS WITH STRUCTURES E.G., EITHER WITH COMMON ELEMENTS SUCH AS A PROPELLANT FUEL TANK OR AS COMPLETELY SEPARATE MODULAR ENGINE OR BOTH	
			• CONTROL AND RELIABILITY OF PROPULSION SYSTEM DURING LEO TO HEO AND LEO TO GEO ORBIT TRANSFER • PROPULSION UNITS UNDER VARYING NON UNIFORM ACCELERATION FROM LOW FREQUENCY STRUCTURAL VIBRATION • EXCITATION OF STRUCTURAL NATURAL FREQUENCIES	SAME
	SUSS A/D IUS	SINGLE POINT	• REMOTE DEPLOYMENT RELIABILITY	

FIGURE 5 INTEGRATED PROPULSION AND STRUCTURES SUBSYSTEM INTERFACE ADVANTAGE MATRIX

STRUCTURAL SUBSYSTEM
(INCREASING SIZE ⟶)

MAIN PROPULSION AND THRUSTER SUBSYSTEMS
(INCREASING THRUST LEVEL ↓)

		DEPLOYABLE	ASSEMBLED
ELECTRIC PROPULSION	SINGLE POINT	• SIMPLIER PROPULSION AND STRUCTURES SUBSYSTEM INTEGRATION AND ANALYSIS	SAME SIZE/THRUST LIMIT
LOW THRUST CHEMICAL	MULTI-POINT	• MULTIPLE LOAD CARRYING PATHS INTO DEPLOYED STRUCTURE • PERFORM MULTIPLE FUNCTIONS, E.G., ORBIT TRANSFER AND ATTITUDE CONTROL, AND STABILIZATION	• INTEGRATION OF PROPULSION SUBSYSTEM WITH EITHER COMMON ELEMENTS OR COMPLETELY SEPARATE MODULES WITH STRUCTURAL SUBSYSTEM SAME ASSEMBLED
SUSS A/D IUS	SINGLE POINT	• EXISTING HARDWARE • RAPID TRANSIT TIME	

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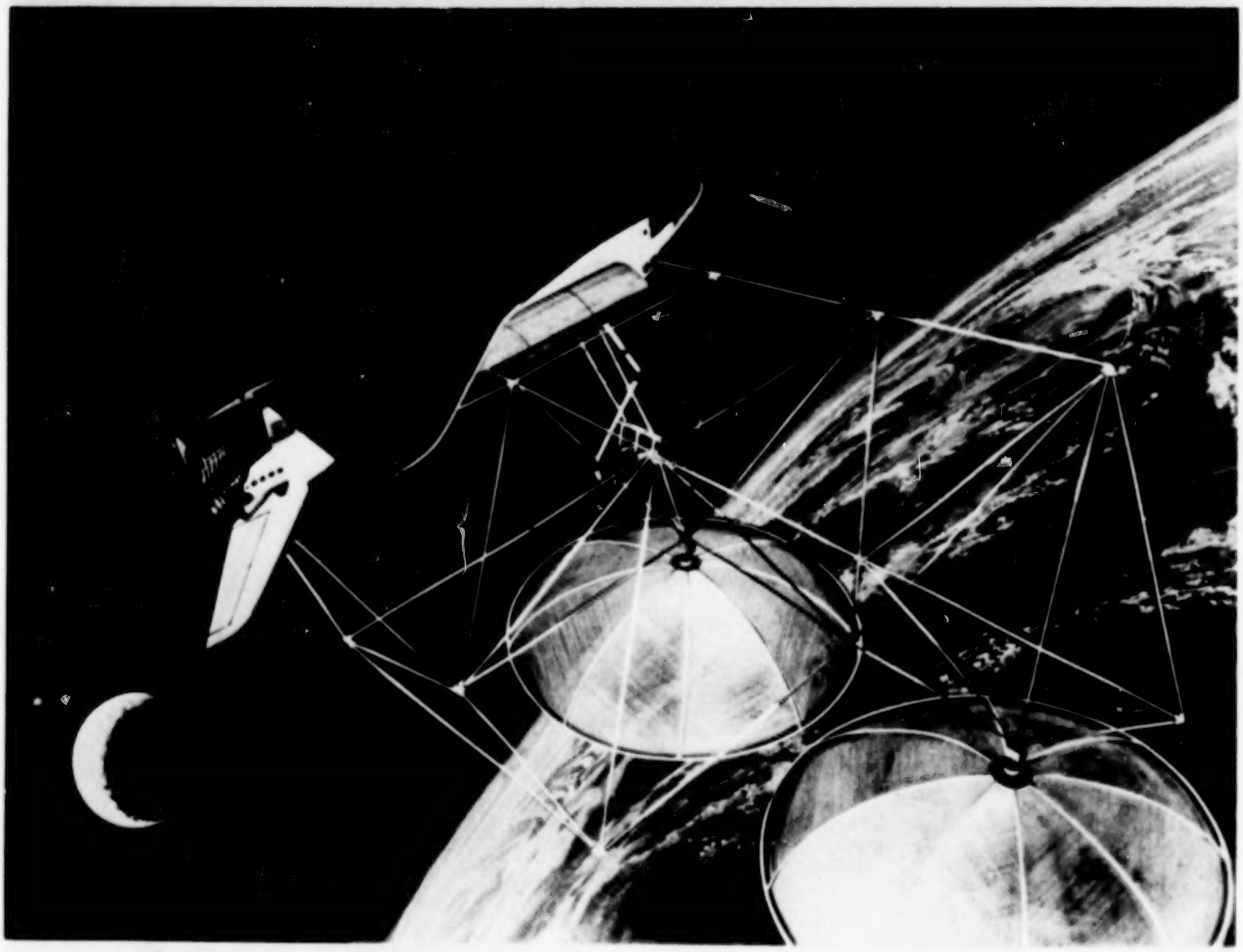
LARGE SPACE SYSTEMS TECHNOLOGY PROGRAM

Robert L. James, Jr.
NASA Langley Research Center

TECHNOLOGY



FOR LARGE SPACE SYSTEMS

In order to provide the capability to design and operate large space systems in the shuttle-era, specific technical challenges must be met as shown on this visual. First, space-configured spacecraft designs must be conceived and developed. Specifically, system designs must be developed which satisfy operational performance requirements and tolerate operational loads. Low environmental and operational loads will lead to lightweight systems. Advanced control systems will be needed to maintain the required attitude and shape control of these lightweight systems.

Secondly, the design and operational requirements of these "large space systems" must be compatible with space shuttle capabilities and limitations. Consequently, the designs must be packageable and assembleable. The packaged system must tolerate the shuttle cargo bay launch environment. Assembly operations must be compatible with capabilities of the shuttle remote manipulator subsystem, the crew, and additional tools and construction aids.

Finally, the overall design of shuttle-era large space systems must be cost effective from the viewpoint of the total mission. Specifically, the packing density must be high. Assembly complexity must be minimized. Selected concepts and techniques should support minimum overall mission cost. For example, while assembly costs may be minimized through the use of deployable elements, the cost of design, fabrication, and testing of these structures might far exceed similar cost elements for erectable concepts. The reliability of on-orbit deployment and/or assembly, and the reliability of the assembled spacecraft will impact overall mission cost and must be considered. Extending the life of components and systems will reduce overall mission costs by reducing the required maintenance and replacement operations. The success in reducing overall mission cost will be a primary factor in the eventual decision to proceed with the development of operational large space systems.

TECHNICAL CHALLENGES OF SHUTTLE-ERA LARGE SPACE SYSTEMS

- THE DEVELOPMENT OF "SPACE - CONFIGURED" SPACECRAFT CONCEPTS
  - DESIGNED TO MEET PERFORMANCE REQUIREMENTS
    - LARGE
    - PRECISION SHAPE
  - DESIGNED FOR THE OPERATIONAL ENVIRONMENT
    - LIGHTWEIGHT
    - ADVANCED CONTROL
- COMPATIBILITY WITH THE SPACE TRANSPORTATION SYSTEM
  - CAPABLE OF BEING PACKAGED WITHIN THE SHUTTLE CARGO BAY
  - CAPABLE OF BEING ASSEMBLED BY THE SHUTTLE WITH TOOLS AND AIDS
- COST EFFECTIVENESS
  - COST - EFFECTIVE PACKAGED VOLUME/WEIGHT
  - COST - EFFECTIVE DEPLOYMENT/ASSEMBLY/CONSTRUCTION TECHNIQUE
  - OVERALL COST EFFECTIVENESS (DESIGN/FABRICATION/TEST/ON-ORBIT ASSEMBLY/OPERATIONS)
  - HIGH RELIABILITY (CONSTRUCTION AND OPERATIONS)
  - LONG-LIF

In order to provide a base of systems technology to enable this new class of spacecraft, the NASA Office of Aeronautics and Space Technology (OAST) established the Large Space Systems Technology (LSST) Program. The multi-Center LSST Program is managed by the NASA Langley Research Center (LaRC). The program is developing fundamental systems technology which will provide a basis for the design of large shuttle-era spacecraft. Ongoing and planned activities will ensure that important initial design choices are made on a sound basis of technical knowledge and experience.

THE LSST PROGRAM

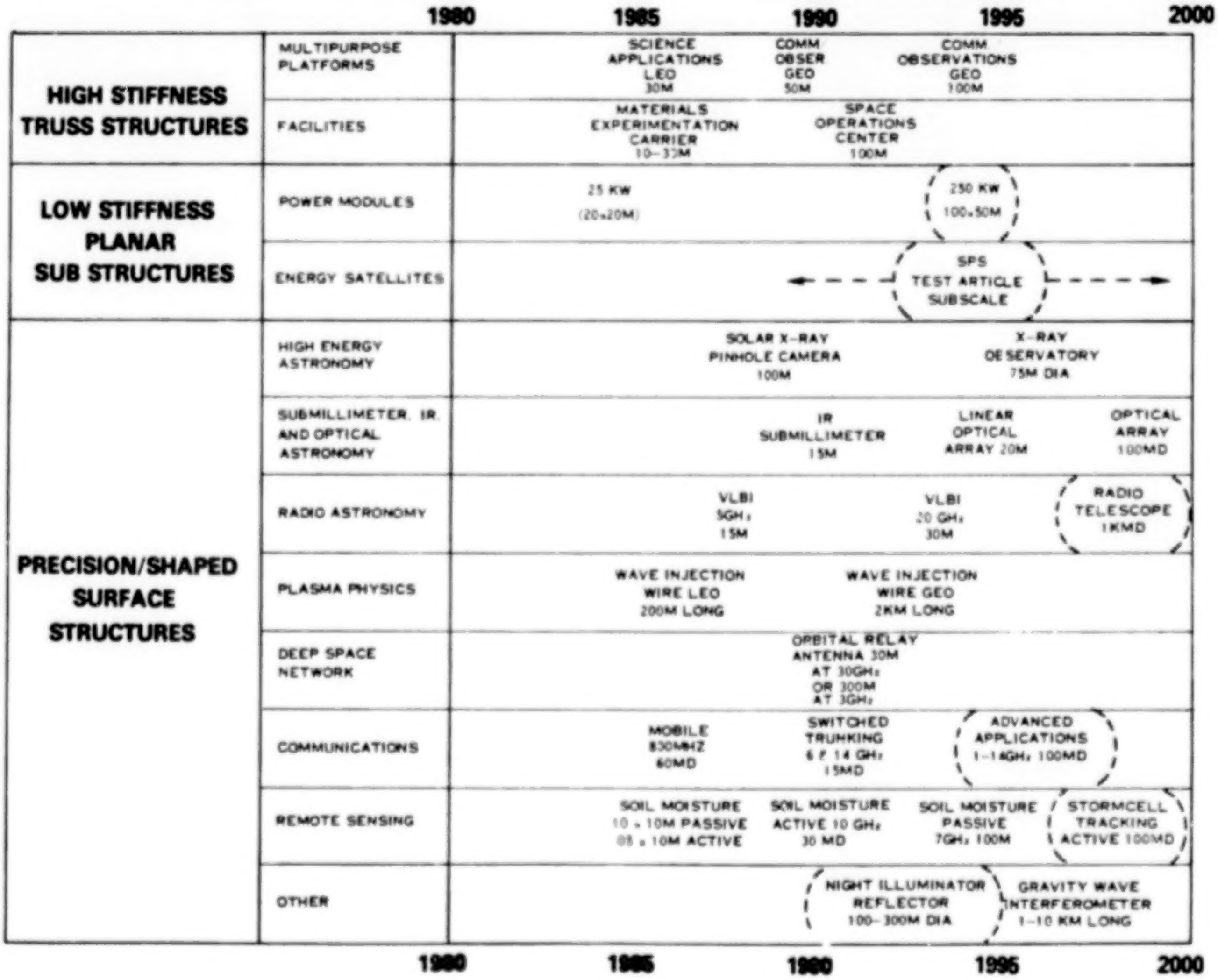
OBJECTIVE:

TO DEVELOP TECHNOLOGY TO ENABLE AND ENHANCE SHUTTLE - COMPATIBLE LARGE SPACE SYSTEMS

- SPONSORING PROGRAM OFFICE:
  - OFFICE OF AERONAUTICS AND SPACE TECHNOLOGY (OAST)
- LEAD CENTER AND PROGRAM MANAGEMENT OFFICE:
  - LANGLEY RESEARCH CENTER
  - LARGE SPACE SYSTEMS TECHNOLOGY (LSST) PROGRAM OFFICE
- PARTICIPATING NASA CENTERS:
  - GODDARD SPACE FLIGHT CENTER
  - JET PROPULSION LABORATORY
  - JOHNSON SPACE CENTER
  - LANGLEY RESEARCH CENTER
  - LEWIS RESEARCH CENTER
  - MARSHALL SPACE FLIGHT CENTER

For the past several years, OAST has periodically surveyed the NASA program offices to identify future space missions which will require large space systems. The results of the most recent survey are shown here. This mission model includes potential missions derived from many sources. Individual missions cover a wide spectrum in level of definition and program office support. However, the compilation gives an overall indication of the strong potential requirements for this class of space vehicle.

POTENTIAL LARGE SPACE SYSTEMS MISSIONS



HIGH STIFFNESS TRUSS STRUCTURES	MULTIPURPOSE PLATFORMS	SCIENCE APPLICATIONS LEO 30M	COMM OBSER GEO 50M	COMM OBSERVATIONS GEO 100M	
	FACILITIES	MATERIALS EXPERIMENTATION CARRIER 10–33M	SPACE OPERATIONS CENTER 100M		
LOW STIFFNESS PLANAR SUB STRUCTURES	POWER MODULES	25 KW (20x20M)	250 KW 100x50M		
	ENERGY SATELLITES	SPS TEST ARTICLE SUBSCALE			
PRECISION/SHAPED SURFACE STRUCTURES	HIGH ENERGY ASTRONOMY	SOLAR X-RAY PINHOLE CAMERA 100M	X-RAY OBSERVATORY 75M DIA		
	SUBMILLIMETER, IR, AND OPTICAL ASTRONOMY	IR SUBMILLIMETER 15M	LINEAR OPTICAL ARRAY 20M	OPTICAL ARRAY 100MD	
	RADIO ASTRONOMY	VLBI 5GHz 15M	VLBI 20 GHz 30M	RADIO TELESCOPE 1KMD	
	PLASMA PHYSICS	WAVE INJECTION WIRE LEO 200M LONG	WAVE INJECTION WIRE GEO 2KM LONG		
	DEEP SPACE NETWORK	ORBITAL RELAY ANTENNA 30M AT 30GHz OR 300M AT 3GHz			
	COMMUNICATIONS	MOBILE 830MHZ 60MD	SWITCHED TRUNKING 6 F 14 GHz 15MD	ADVANCED APPLICATIONS 1–14GHz 100MD	
	REMOTE SENSING	SOIL MOISTURE 10 x 10M PASSIVE 05 x 10M ACTIVE	SOIL MOISTURE ACTIVE 10 GHz 30 MD	SOIL MOISTURE PASSIVE 7GHz 100M	STORMCELL TRACKING ACTIVE 100MD
	OTHER	NIGHT ILLUMINATOR REFLECTOR 100–300M DIA	GRAVITY WAVE INTERFEROMETER 1–10 KM LONG		

The identified potential missions fall primarily in two classes: large antennas and platforms. In order to provide an integrating focus to the technology development, the LSST Program has selected a set of reference missions which collectively represent the technology challenges. These missions are studied to define technology requirements and to identify subsystem interfaces.

REFERENCE MISSIONS OF THE LSST PROGRAM

- LARGE ANTENNAS
    - Mobile communications
        - 60 - 100 m (180 - 300 ft)
        - 0.8 - 14.0 GHz ($\lambda/20$ surface accuracy)
    - Very long baseline interferometer (VLBI)
        - 40 - 80 m (120 - 240 ft)
        - 1.4 - 14.0 GHz ($\lambda/10$ surface accuracy)
    - Orbiting deep space relay station (ODSRS)
        - 20 - 50 m (60 - 150 ft)
        - 3.0 - 30.0 GHz ($\lambda/30$ surface accuracy)
    - Radiometers
        - 30 - 100 m (90 - 300 ft)
        - 1.4 - 10.0 GHz ($\lambda/50$ surface accuracy)
- PLATFORMS
    - Advanced science/applications platform
    - Operational Geosynchronous communications/observations platform
    - Satellite Power System (SPS) engineering test article
    - Space Operations Center (SOC)

The LSST Program is currently subdivided into the elements shown. These elements comprise the primary technology needs of near-term shuttle-era large space structural systems. Included are the structural systems and related technologies. Program activities are also undertaken to define the interfaces of the other subsystems to the structure.

ELEMENTS OF THE LSST PROGRAM

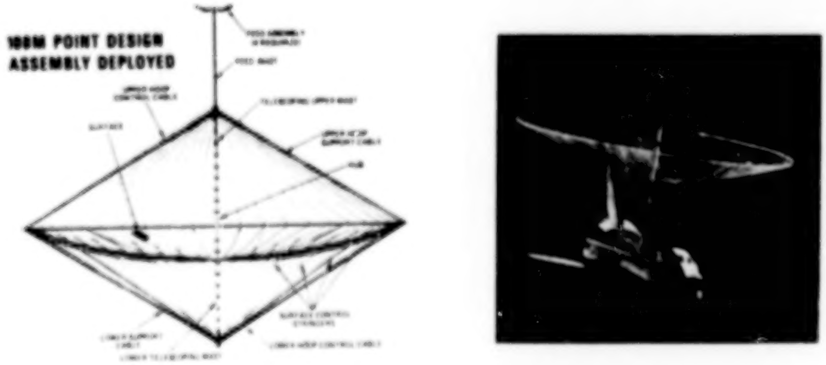
- PROGRAM PLANNING, INTEGRATION AND MANAGEMENT
  - PROGRAM MANAGEMENT
  - SYSTEM REQUIREMENTS AND INTERFACE DEFINITION
  - FLIGHT EXPERIMENT DEFINITION
- ANTENNAS
  - MAYPOLE (HOOP/COLUMN) CONCEPT DEVELOPMENT
  - OFFSET WRAP-RIB CONCEPT DEVELOPMENT
  - ELECTROMAGNETIC ANALYSIS
- SPACE PLATFORMS
  - DEPLOYABLE SYSTEMS
  - ASSEMBLY METHODS
  - MODULAR CONTROL SYSTEMS
- ASSEMBLY EQUIPMENT AND DEVICES
  - LARGE PLATFORM ASSEMBLER TECHNOLOGY
  - ASSEMBLY AND CONSTRUCTION EQUIPMENT
- SURFACE SENSORS AND CONTROL
  - STRUCTURAL MEASUREMENT SYSTEMS
  - ELECTROSTATIC SHAPE CONTROL
  - ACTIVE SHAPE AND ALIGNMENT SENSOR AND ACTUATOR CONCEPTS
- CONTROL AND STABILIZATION
  - LARGE SPACE SYSTEMS CONTROL
- ANALYSIS AND DESIGN SYSTEMS
  - INTEGRATED ANALYSIS AND DESIGN

The LSST antenna technology program has as its objective the development of the antenna technology required to support the large antenna reference missions.

The offset wrap-rib antenna concept development activity will develop antenna technology for classes of applications which require large unblocked apertures of up to 1000 feet (300 m). Development activities will include definitization of the antenna design (surface quality, weight, deployable feed support structure), definition of scaling laws, development of structural and thermal analysis techniques, characterization of surface adjustment techniques, development of a feed support structure, the development and evaluation of critical components, and the development of cost and performance models. At the present time, design requirements have been determined and the reflector configuration optimized. Currently, the feed support structure is being optimized, and surface adjustment techniques are under evaluation. In the near future, fabrication of critical components for a 180 foot (55 m) model will be initiated.

THE TECHNOLOGY FOR LARGE ANTENNAS

OFFSET WRAP-RIB CONCEPT

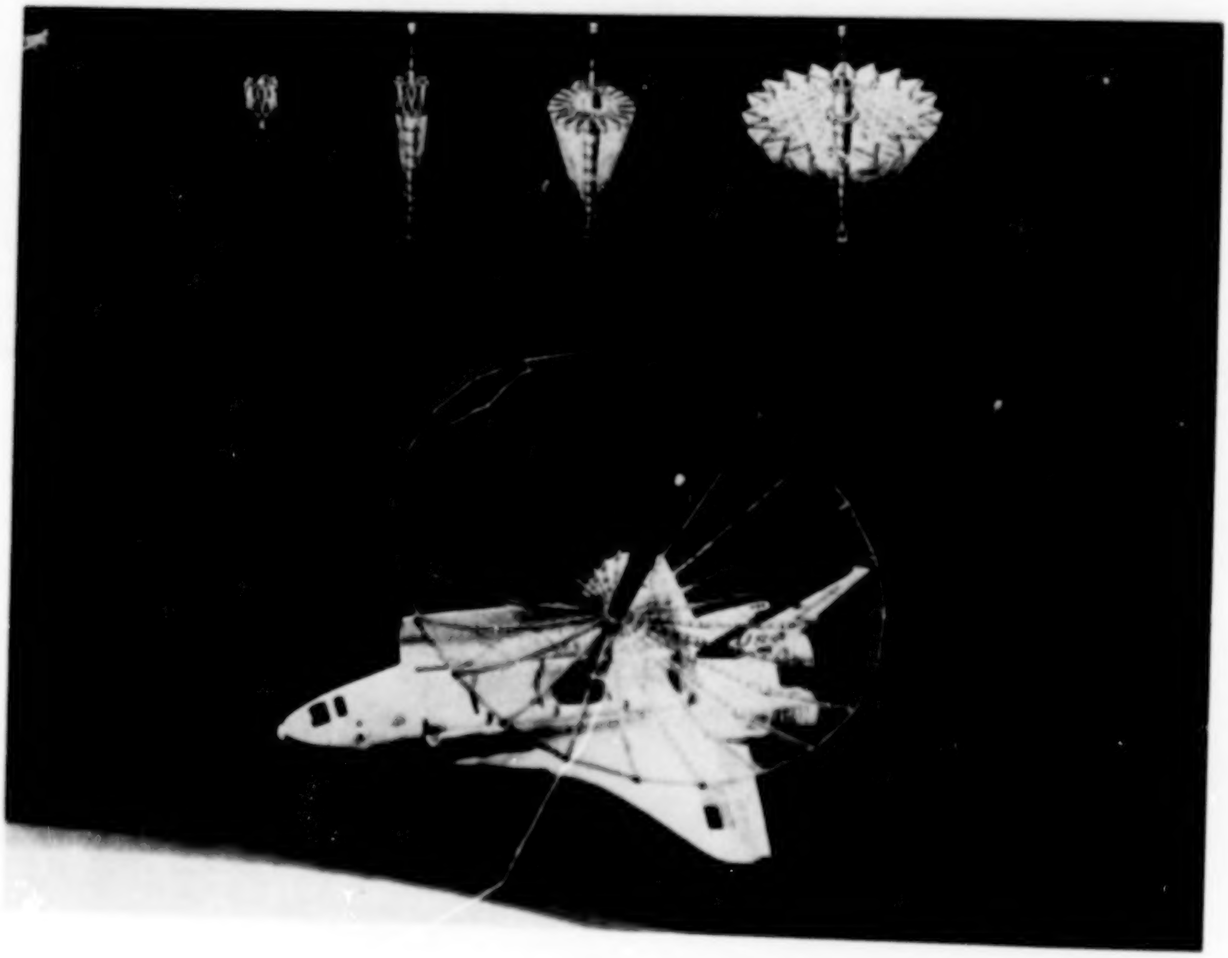


MAYPOLE (HOOP/COLUMN) CONCEPT

The objective of the Maypole (hoop-column) antenna concept development is to structurally characterize this antenna concept and to determine its performance through tests and analyses. Primary activities are to determine surface quality as a function of size, to develop structural and thermal analysis techniques, to define the dynamic behavior of the antenna during deployment, and to define ground-test requirements. In addition, the activity will define and evaluate surface adjustment techniques, define scaling laws, and develop cost models. Currently, the antenna configuration has been defined, and a point design of a 300-foot (100-m) antenna is nearing completion. The end product is expected to be a data base which will permit estimates of performance and cost for Maypole (hoop/column) antennas up to 1000 feet (300 m) in diameter.

Also, included in the antenna technology program is the development of analysis techniques for predicting electromagnetic performance of a broad class of large reflectors. These techniques will show specific effects of surface errors and distortions and their correlation and distribution on antenna performance.

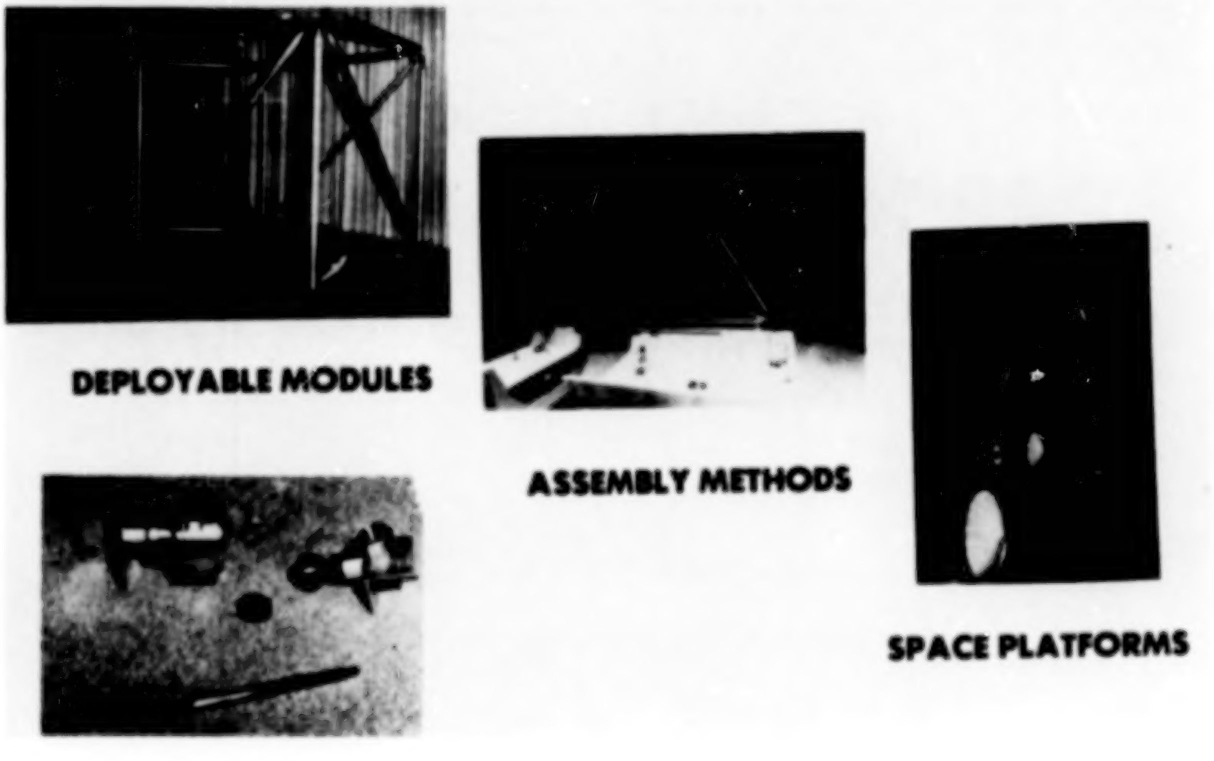
DEPLOYABLE ANTENNA



An important class of structural concepts are the deployable trusses. This structural concept is attractive for space construction because major subassemblies can be completely fabricated and functionally checked out on the ground. The deployable concept is also attractive in that it minimizes the time required for on-orbit construction and checkout. However, deployable structural concepts present designers with a number of difficult technical challenges. Compared to other concepts, deployable structures have a low packaging efficiency. Therefore, optimum folding concepts must be developed. The joint concept has a critical impact on reliability of the deployment process. The degree of joint rigidization following deployment can strongly effect the structural dynamic behavior. The overall reliability of the system depends on the development of reliable deployment concepts and mechanisms. Prediction of deployment dynamics requires the development of new models and test data for validation. Finally, the structural concept must be functionally useful. Therefore, as the concepts are developed, it will be necessary to include provisions for utility distribution and subsystem integration.

The overall objective of the space platform element of the LSST Program is to develop the technology needed to design, fabricate, package, and automatically deploy structurally efficient linear or area platform structures. Specific activities will include the concept definition of several alternative deployable modules. The mechanisms necessary for the implementation of the concepts will be designed, fabricated, and tested. Currently, a double-fold concept has been designed and partially tested. Full-scale module-to-module couplings have been designed and tested. A 1/2-scale model of the deployable module has been fabricated and tested. Three full-scale 18-foot (5-m) modules are under fabrication for use in deployment and assembly tests. The modules have been designed to accommodate assembly test in a neutral buoyancy facility.

THE TECHNOLOGY FOR LARGE SPACE PLATFORMS



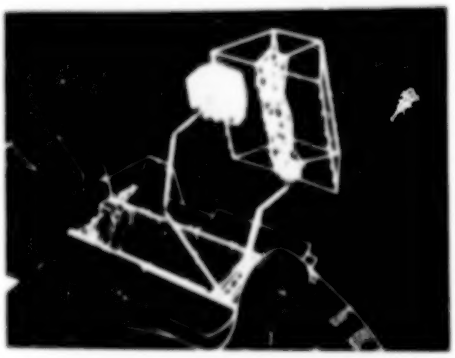
The potential ability of the space shuttle to assist in the on-orbit assembly of packaged spacecraft is a fundamental consideration in development of this new class of spacecraft. The LSST Program is conducting activities which will develop and evaluate efficient packaging and assembly techniques. The planned tasks will consider assembly techniques ranging from manual to fully automated. Analyses and simulations will be performed to define the capabilities and limitations of the various techniques. The experimental results will provide data on which to base the selection and development of cost-effective assembly techniques.

Primary initial tests have addressed the capabilities and limitations of extravehicular activity (EVA) for assembly operations (previous graphic). Of the various techniques, EVA is considered to be a base of reference. This technique is the only method on which any space experience exists. Assembly by EVA is shown to be very time-consuming and relatively inefficient. However, EVA assist may be very effective for specific operations. In fact, on the basis of past space experience, some operations may not be possible without EVA assist.

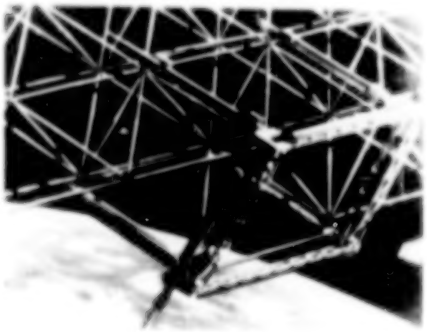
A comprehensive series of assembly tests are currently underway in the Neutral Buoyancy Facility at the Marshall Space Flight Center. This facility includes a cylindrical water tank with a depth of 40 feet (12 m). Tests conducted in the tank simulate operations in zero gravity. The objectives of current experimental testing are to develop manual assembly techniques, identify fundamental requirements for multipurpose assembly aids, evaluate various techniques for the assembly of elements and subassemblies, define assembly time lines for the various techniques, and identify fundamental limitations of this assembly method. Testing has included the manual connection of an electrical connector designed for quick assembly, assembly of a tetrahedral cell with various member lengths and joint concepts, and the attachment of a simulated equipment or subsystems module. Extensive testing was performed on the assembly of the tetrahedral cell. Experimental tests were performed using 30-foot (9-m) and 18-foot (5-m) strut lengths, using both the snap-lock and ball-and-socket joint concepts. These tests have shown the feasibility of manual assembly. They have also demonstrated the critical importance of joint design and the need for basic assembly aids.

Machine-aided assembly techniques appear to offer many advantages. The concepts offer the potential for automation which could significantly reduce assembly time. Activities planned within the LSST Program will develop concepts for RMS-aided assembly techniques and define the requirements for special end-effectors and assembly aids. Assembly concepts will be evaluated to experimentally define time lines and fundamental limitations of the approach. Automated assembly of space structures is an advanced concept which may be the only practical approach for the assembly of very large systems.

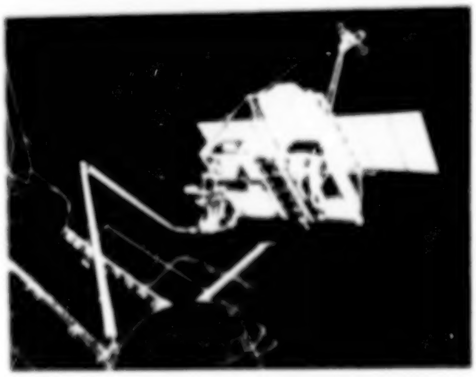
ASSEMBLY TECHNOLOGY



AUTOMATED ASSEMBLY



ASSEMBLY AND CONSTRUCTION EQUIPMENT



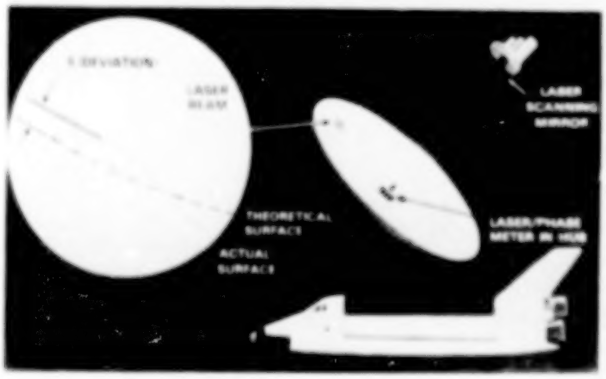
Large systems in space will require an ability to precisely determine and statically control surface contours. Two surface measurement systems are currently under test and evaluation for application to large antenna concepts. Evaluations of breadboard units will be completed this year. The design objective is a surface measurement precision of 0.1 mm at a distance of 500 feet (150 m).

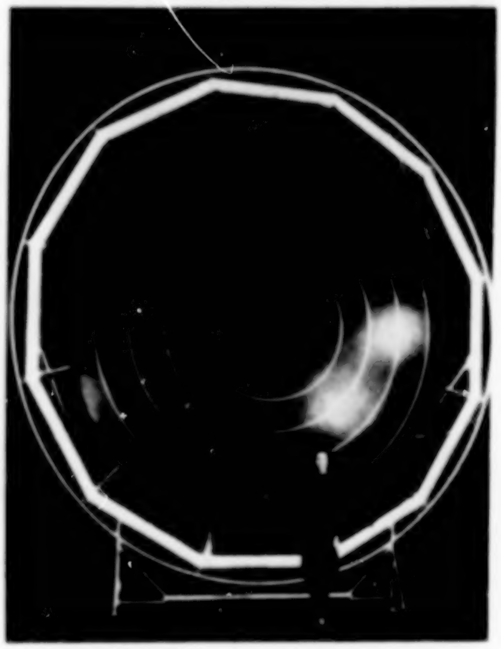
Effective surface control actuators for large systems will greatly improve the ability to compensate for alignment errors and operational deflections. Concepts for surface control of the wrap-rib and hoop/column antennas have been defined and are under evaluation. These systems may be required to compensate for environmentally induced deflections of the surface for very large systems.

Electrostatic shape control of a membrane is also under study. Objectives of this activity are to determine the feasibility of using electrostatic forces to control membrane surfaces, the selection of suitable materials, quantify the surface control capability of the technique, and to determine the effects of spacecraft charging. A 16-foot (5-m) model has been fabricated and surface-shaping tests initiated. Initial tests will be for the purpose of membrane material evaluation.

SURFACE SENSING AND CONTROL

SURFACE SENSING





ELECTROSTATIC SURFACE CONTROL

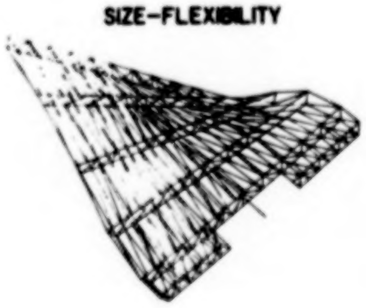
The unique structural characteristics of efficient space-configured spacecraft place a new requirement on control technology. Future large flexible antennas and space platforms will require precise attitude and shape control to satisfy mission requirements. New capabilities, such as active figure control, may be required to provide accurate surface contours and vibration suppression to ensure long-term structural integrity. Analyses have shown that these future structural systems will dynamically react with the control systems performance capability and potentially result in unstable control/structures interactions. Advanced control concepts tolerant of model errors with the capability to handle many interactive degrees of freedom must be developed to permit these large systems to satisfy performance requirements. The LSST Program supports a broad controls technology activity to address these needs.

Potential control problems associated with large space structures result from model inadequancies, including parameter uncertainty and variability, unmodeled nonlinearities, unmodeled disturbances, model truncation, and from interactions between the structure and the control systems. The LSST Program is sponsoring tasks at the Jet Propulsion Laboratory (JPL) which address these questions.

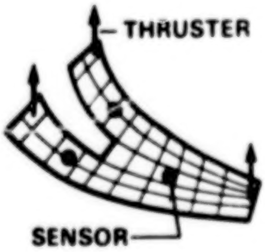
JPL and Purdue University are investigating the reduction of model order to minimize on-board computations and implementation complexity. To date, the investigators have defined the stability, controllability, and observability of dynamical systems established a finite element model of a generic configuration, and performed a modal analysis. The tasks are expected to provide model-order reduction methods for reduced-order controller design.

JPL is also attempting to design estimators capable of on-board detection of deficiencies in large structural dynamical models. This work is an extension of experience gained in state estimation and control of planetary spacecraft with flexible appendages. Finally, work is continuing to develop distributed control concepts. At JPL, a local distributed control system has been designed for beam-like structures. This technique is simpler to implement because of reduced dimensionality. Methods for static-shape estimation and sensor and actuator placement have also been studied. These studies are of fundamental importance and have wide potential application.

CONTROL AND STABILIZATION



- **DISTRIBUTED CONTROL**
- **MODEL ORDER REDUCTION**
- **MODEL ERROR ESTIMATION**



SIZE-COMPLEXITY

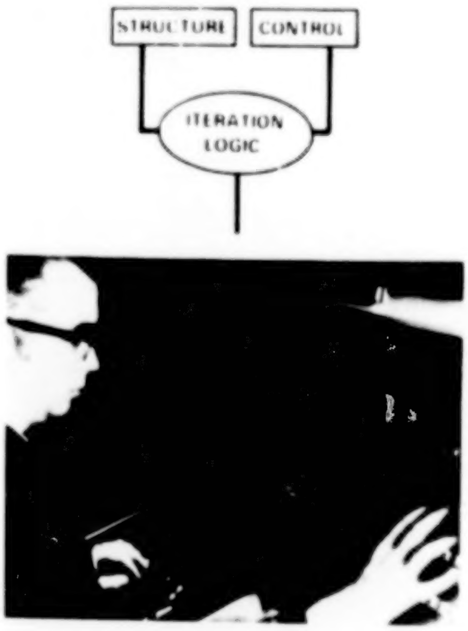
The capability to accurately perform the structural, thermal, and control analysis of a spacecraft in an efficient manner is important to spacecraft designers. Problems of interpretation and inefficiency frequently result from an absence of interaction between the various disciplines. These problems become more acute as the structural size and flexibility increases.

The LSST Program is sponsoring the development of an interactive analysis program at the Goddard Space Flight Center. The computer program will couple the thermal, structures, and control analysis. Early emphasis will be on the practical condensation of transient thermal analysis models and on improved technique for analyzing sampled data control systems. The end product of these tasks is expected to be an operational integrated analysis computer program suitable for preliminary design.

INTEGRATED ANALYSIS AND DESIGN

OBJECTIVE

- **PROVIDE EFFICIENT CAPABILITY TO COUPLE STRUCTURAL, THERMAL, AND CONTROL ANALYSES**



PRELIMINARY DESIGN

CONCLUDING REMARKS

- LARGE SIZE WILL MAKE SIGNIFICANT CONTRIBUTIONS TO THE PERFORMANCE AND UTILITY OF SPACE SYSTEMS
- SHUTTLE CAPABILITIES WILL ENABLE THESE SYSTEMS
- TECHNOLOGY ADVANCEMENTS ARE NEEDED TO REDUCE THE COST AND RISK
- THE LSST PROGRAM IS PROVIDING TECHNOLOGY WHICH WILL ACCELERATE THE TECHNICAL AND ECONOMIC FEASIBILITY

ELECTRIC PROPULSION TECHNOLOGY

Robert C. Finke
NASA Lewis Research Center

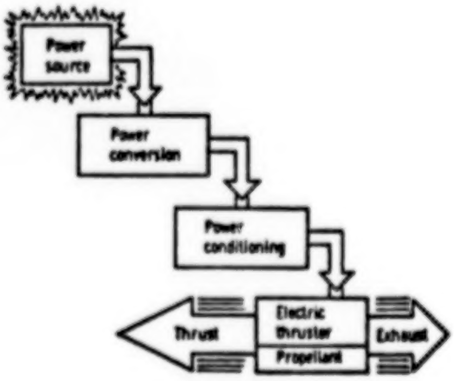
Propulsion systems can be classified into two basic categories:

I. Endogenous; which use energy stored within the propellants to create thrust. Solid rockets, liquid rockets, cold gas systems, etc. are all well known examples of endogenous systems.

II. Exogenous; in which the energy is supplied to the propellant from an outside power source. All electric propulsion systems are exogenous although some like electrically augmented hydrazine are a combination of the two.

WHAT IS ELECTRIC PROPULSION?

ELECTRIC PROPULSION IS A PROCESS IN WHICH ELECTRICAL ENERGY IS USED TO ACCELERATE A PROPELLANT TO HIGH VELOCITY CREATING THRUST.


Power source
Power conversion
Power conditioning
Electric thruster
Propellant
Thrust
Exhaust

The most significant advantage of an exogenous system is that if external energy is available for accelerating a propellant, the resulting specific impulse and total impulse can be greatly in excess of that that can be stored in an endogenous device. Thus an ion thruster system with an I_{sp} of 3000 sec would require 2000 kg of propellant as compared to 15,000 kg of propellant for a Centaur with equivalent total impulse. The dry weights of the two systems are also similar, resulting in a significant advantage for the ion thruster system.

Electric propulsion devices are inherently low thrust devices. A cluster of ten 30-cm thruster systems provides a 0.3 pound thrust to the system for up to 15,000 hours of operation. The low level continuous thrusting characteristic of Electric Propulsion allows very fragile large space structures to be transported by these class of propulsion systems, assembled, from LEO to GEO.

In addition, since propellant is a very small fraction of overall system mass, weight growth of the payload during the construction phase of the project can be accommodated by thrusting for a longer period of time; increased mass then merely requires longer trip times.

CHARACTERISTICS OF ELECTRIC PROPULSION

- HIGH SPECIFIC IMPULSE
  - LARGE TOTAL IMPULSE FOR LOW MASS
  - MINIMUM PROPELLANT REQUIREMENTS
- LOW THRUST
  - LOW "G" LOADING ON SPACECRAFT STRUCTURES
  - PRECISION POINTING CAPABILITY PROVIDED
- HIGH POWER REQUIRED
  - EXCELLENT MATCH WITH HIGH POWER PAYLOADS
- ORBIT TRANSFER TIME/PAYLOAD TRADE AVAILABLE
- COMPATIBLE WITH LONG TERM SPACE STORAGE/OPERATIONS

There are three generic classes of electric propulsion devices, all of which are capable of high impulse. The electrostatic devices, in particular are capable of a wide range of specific impulses.

- ELECTROTHERMAL

  In the electrothermal rocket electric power is used to heat the propellant to a high temperature. The heating may be accomplished by producing an electric discharge through the propellant gas (arcjet) or by flowing the propellant gas over surfaces heated with electricity (resistojet).

  The electrothermal rocket is similar in some respects to the chemical rocket. Although there is no combustion, the propellant gas is heated to high temperatures and expanded through a nozzle to produce thrust. This rocket can achieve propellant exhaust velocities higher than those of chemical rockets because the energy added to the gas molecules may be larger than the energy available from combustion. Material failure at high temperature, however, places a practical upper limit on the amount of energy that can be added to the propellant. Other factors, such as breakup, or dissociation, of the propellant gas molecules, which absorbs energy without raising gas temperature much, also limit the exhaust velocity.

- ELECTROMAGNETIC

  The second general type of engine is the electromagnetic thruster, often called the plasma thruster. In this thruster, the propellant gas is ionized to form a plasma, which is then accelerated rearward by electric and magnetic fields.

  In a plasma, the electrons and the ions are swirling about in a random manner much like atoms in a gas. The plasma can conduct electric current just as a copper wire can conduct current. It is this conductivity that makes possible accelerating the plasma electrically and magnetically. When an electric current is made to pass through a plasma in the presence of a magnetic field, a force is exerted on the plasma. Because of this force, the plasma is accelerated rearward. Thus, a plasma thruster is quite similar to an electric motor with the plasma replacing the moving rotor.

- ELECTROSTATIC

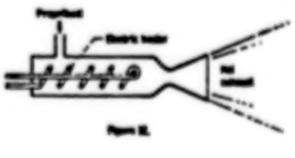
  The third type of electric rocket engine is the electrostatic thruster. (Best known of this type is the ion thruster or ion engine.) As in the plasma thruster, propellant atoms are ionized by removing an electron from each atom. In the electrostatic thruster, however, the electrons are removed form the ionization region at the same rate as ions are accelerated rearward.

  The most successful electrostatic thruster presently available is an electron-bombardment thruster conceived and developed at the NASA-Lewis Research Center. This thruster operates as follows. When heated, the propellant evaporates and forms a vapor, which is fed into the thruster discharge chamber. In the chamber, electrons are knocked out of many of the propellant atoms to form ions. This ionization is accomplished in a gentle electric discharge wherein electrons in the discharge hit electrons in the atom and displace them from the structure of the atom. The electrons and the ions form a plasma in the ionization chamber. The electric field between the screen and the accelerator draws ions from the plasma. These ions are then accelerated out through many small holes in the screen and accelerator electrode.

WHY - ELECTRIC PROPULSION?

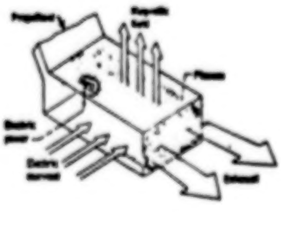
- CHEMICAL ENERGY IS LIMITED TO SPECIFIC IMPULSES 500 SEC.

- ELECTRIC PROPULSION IS CAPABLE OF A BROAD RANGE OF SPECIFIC IMPULSE.



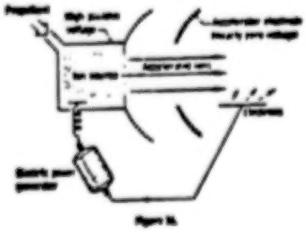
- **ELECTROTHERMAL**

  350 - 1200 SEC



- **ELECTROMAGNETIC**

  200 - 2000 SEC



- **ELECTROSTATIC**

  1500 - 100,000 SEC

ELECTROSTATIC

Applications of electric propulsion are many and varied. Electrostatic thrusters with their capability for a broad range of specific impulse and ability to scale and throttle over a wide thrust range, are suitable for primary propulsion applications for planetary and earth orbital missions and as auxiliary propulsion devices for attitude control and stationkeeping of geostationary spacecraft. Operation with a wide variety of propellants has been demonstrated from the heavy metals such as mercury or cesium to gases such as argon, xenon, neon and nitrogen.

With an electrostatic thruster system, it is possible to tailor the thruster systems very closely to the application.

ELECTROMAGNETIC

Electromagnetic thruster systems offer the promise of reduced complexity of power systems and high thrust density. In general they are plasma devices and are thus self-neutralizing eliminating the need for a neutralizer system.

One sub-class of electromagnetic thruster can accelerate solid project files. This class represented by the rail gun and mass driver may make possible the direct launch of payloads from earth to space, or the augmentation of booster capabilities via an electric catapult device.

ELECTROTHERMAL

Electrothermal thrusters most resemble the classical chemical rocket. Many such as electrically augmented catalytic hydrazine are techniques to increase the I_{sp} from chemical reaction by the addition of electric power. Others, such as the free radical propulsion concept represent a way to use electrical energy to dissociate H_2 and utilize the high temperatures of recombination to obtain high I_{sp} at high thrusts.

	PLANETARY	ADVANCED PLANETARY	ORBIT TRANSFER	DIRECT LAUNCH	EXTRA SOLAR SYSTEM	ON-ORBIT OPS.
ELECTROSTATIC						
BASELINE HG	●		●			●
ADVANCED HG		●	●			●
INERT GAS			●			●
ELECTROMAGNETIC						
M P D		●	●			
MASS DRIVER			●	●	●	
RAIL			●	●	●	
ELECTROTHERMAL						
FREE RADICAL		●	●			
RESISTOJET			●			●

LSS PROPULSION REQUIREMENTS

Scenarios presently being considered for Large Space Structures (LSS) will require technology advancements to enhance the capabilities of existing propulsion systems, both for orbit raising and for on orbit applications. Almost all studies of LSS have indicated that for balancing out solar pressure, configuration control and maintaining required pointing accuracy will require propulsion systems with a specific impulse well beyond that obtainable from chemical systems.

In addition, the cost of transporting heavy, high volume chemical propellant systems from earth to orbit will become prohibititve as system requirements increase.

In an attempt to minimize mass to orbit, LSS will be designed to be relatively fragile structurally. Large impulsive loads could literally destroy the LSS. In this respect, electric propulsion systems are well matched to LSS since accelerations produced by proposed and existing electric propulsion systems suitable for LSS are all less than 10^{-3} g's.

LSS PROPULSION REQUIREMENTS

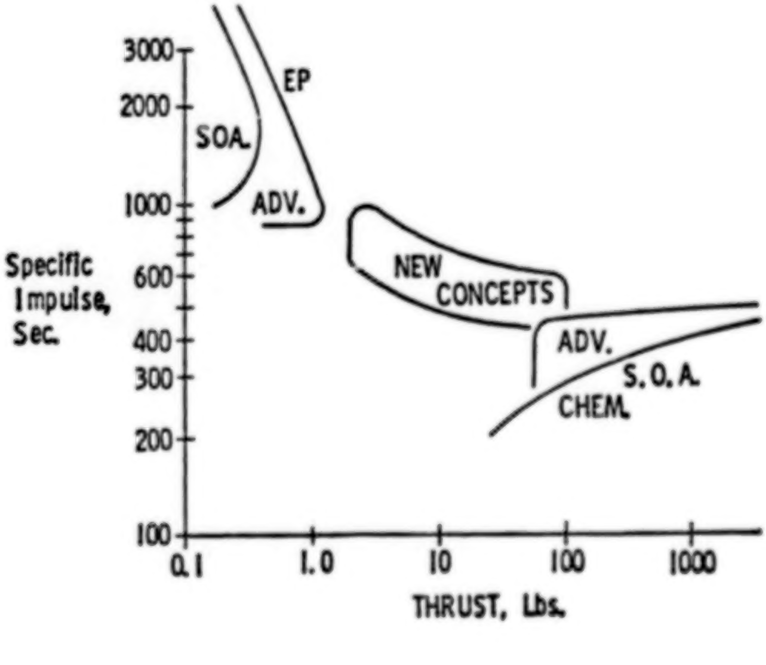
- TOTAL IMPULSE AND MISSION LIFE REQUIREMENTS WILL EXCEED PRESENT CAPABILITIES
- LIFE/CYCLE COSTS A MAJOR FACTOR
  - MINIMIZE TOTAL SYSTEM MASS REQUIRED IN SPACE
  - MINIMIZE PROPULSION SYSTEM VOLUME/LENGTH/MASS/COST
  - MAXIMIZE INHERITANCE AND UTILITY OF SYSTEM CONCEPTS(s)
- MANY LSS ORBIT TRANSFER AND ON-ORBIT APPLICATIONS REQUIRE LOW ACCELERATION
- PROPELLANT AVAILABILITY AND ECOLOGICAL CONCERNS.

PROPULSION CONCEPTS

The advanced chemical propulsion program is structured towards the development of technology for high I_{sp}, low thrust, long life thruster systems suitable for taking payloads from LEO to GEO orbit. The Advanced Electric Propulsion program is directed towards lowering the specific impulse and increasing the thrust per unit of ion thruster systems. In addition, electrothermal and electromagnetic propulsion technologies are being developed to attempt to fill the gap between the conventional ion thruster and chemical rocket systems.

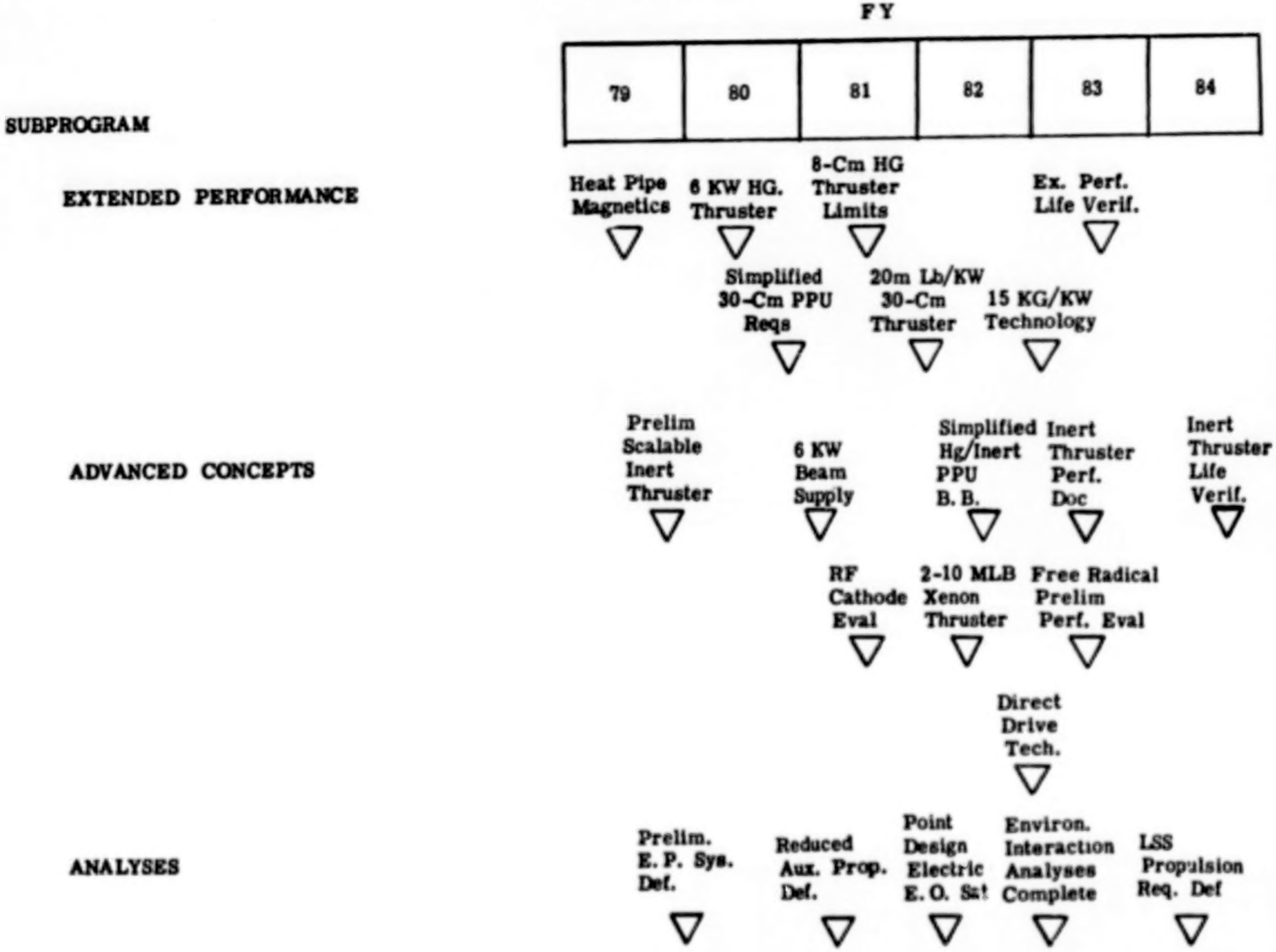
Most of these new concepts are exagenous and are represented by rail accelerators, ablative teflon thrusters, MPD arcs, Free Radicals, etc. Endogenous systems such as metalic hydrogen offer great promise and are also being pursued.

PROPULSION CONCEPTS



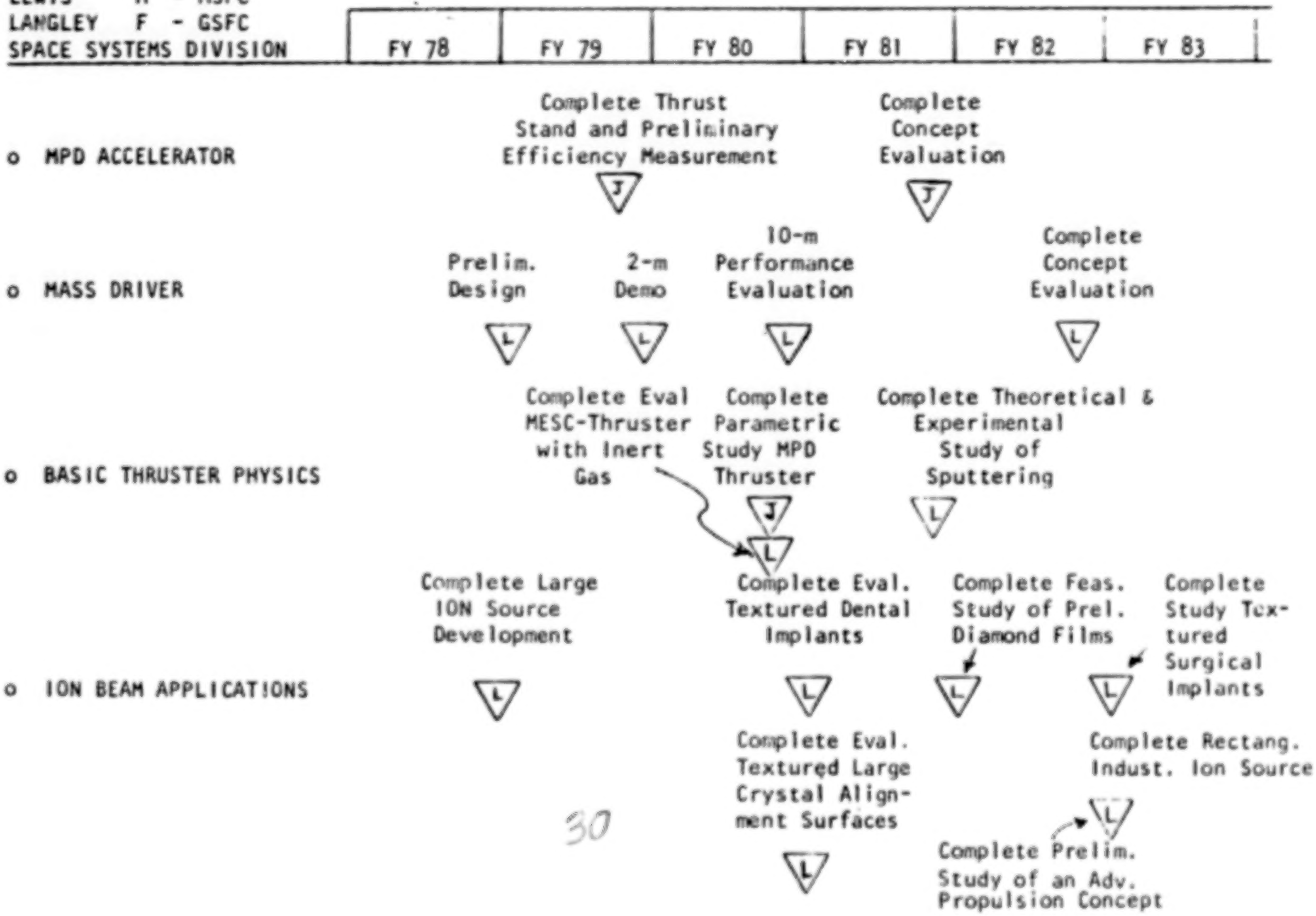
ELECTRIC PROPULSION TECHNOLOGY

506 - 55 - 22



ELECTRIC PROPULSION - RESEARCH AND ADVANCED CONCEPTS

J - JPL Js - JSC
L - LEWIS M - MSFC
La- LANGLEY F - GSFC

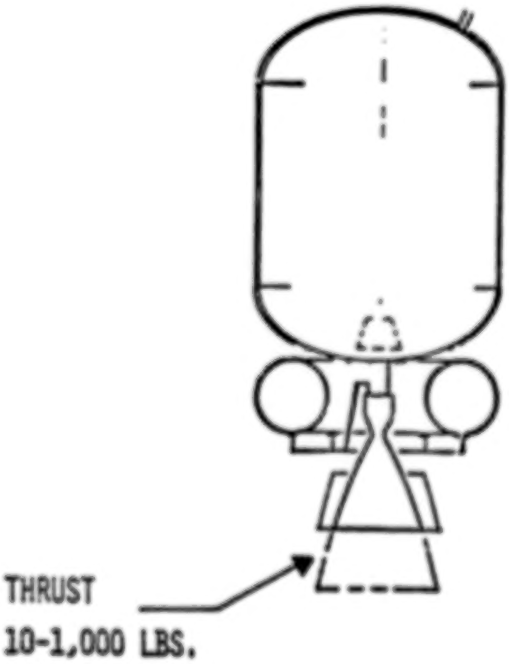


CHEMICAL PROPULSION TECHNOLOGY

Richard J. Priem
NASA Lewis Research Center

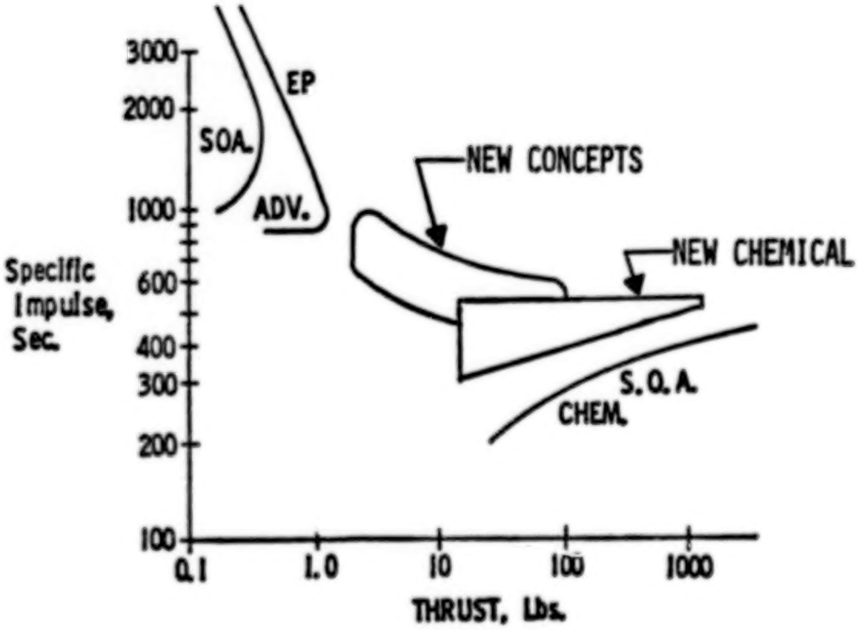
NASA's Low Thrust Liquid Chemical Propulsion Program is represented in the following charts. They have been used in presentations to several of the NASA Overview Committees in the past couple of months and are in a program plan that contains most of this information, so they represent an overall view of the chemical propulsion technology program. This presentation pertains to thrust system technology in the ten to thousand lb. thrust range. This chart schematically shows the elements of the propulsion system, with tanks, structures, and engines included in the program.

LOW THRUST CHEMICAL PROPULSION TECHNOLOGY PROGRAM



This chart shows that the new chemical program that we are talking about is in the ten to thousand lb. thrust range and a specific impulse which is close to 500 sec. The state-of-the-art drops off very rapidly in the low thrust range. This is why we are speaking of a dedicated thrust system in the low thrust range. There are other programs in the thousand lbs. and higher thrust range that are used for orbital transfer. I am not discussing that today.

PROPULSION SYSTEMS FOR LSS



The objective of the program, shown in this chart, is a technology program as Dick Carlisle mentioned before: We want to provide the tools, data, and analyses to allow propulsion system designers and people that do mission studies to optimize the actual system. We also need to develop new techniques that are required for this low thrust system, including throttling of the chambers, how to cool very small chambers, pumps and packaging of the complete system. The program also has to demonstrate the technology readiness, both in the components and possibly in the total propulsion system.

OBJECTIVES OF LTCP PROGRAM

TECHNOLOGY PROGRAM THAT:

- PROVIDES TOOLS (DATA, ANALYSIS, DESIGN PROCEDURES) TO DEFINE PROPULSION SYSTEM PERFORMANCE, WEIGHT, SIZE, ETC. IN TERMS OF ENGINE DESIGN VARIABLE (THRUST, PRESSURE, ETC.)
- DEVELOPS NEW TECHNIQUES NEEDED FOR LSS MISSIONS (THROTTLING, COOLING, PUMPS, PACKAGING, ETC.)
- DEMONSTRATES TECHNOLOGY READINESS (COMPONENTS AND PROPULSION SYSTEM)

The elements of the program are shown in this chart. The fundamental technologies are studies to establish what the engine requirements are. Cooling studies are listed, because cooling is a severe problem, especially at low thrust. We think that high pressure might be required, so we have included pumps, bearings, and seals. Also included are throttling concepts. In the components and engine systems area, we have to design and test these components to demonstrate that they are truly feasible and that the critical technology is available. We also need simulation tests of the engine systems for the most difficult technology. This is where we are not sure how far we have to go (how far we have to get into simulation testing as part of the program). The last step would be for a breadboard system to demonstrate that the complete technology is ready for a full system development. Again, we are not sure at this time whether a breadboard system would be required, but have included it in the program.

The key issue that we see in the low thrust chemical propulsion is high performance of cooled low thrust engines. You have already seen that we have achieved low performance, low I_{sp} down in these low thrust ranges. Now we must demonstrate high performance and long life, which requires cooling. We believe this will require small cryogenic pumps, and they are not available in the state-of-the-art. Multiple starts appears to be a requirement for perhaps ten starts and shutdowns, with a slow ramp such that the structure is not damaged by a sudden change in acceleration. Thrust variation could be 4 to 1 in flight so that constant g's are maintained as propellant is used up. For different missions, it is possible that a thrust range of 20 to 1 would be needed. Very long life is required. At very low thrust levels, a hundred hours of engine firing time is needed to complete a mission. We also have to improve the system weight and size. The final item is the selection of propellants, because defferent propellant systems have different characteristics that might be desirable for different missions. These are the key issues as we see them right now.

KEY TECHNOLOGY ISSUES FOR
LOW THRUST CHEMICAL PROPULSION

- High Performance of Cooled - Low Thrust Engines
- Small Cryogenic Pumps
- Multiple Starts - Shutdowns (10) with Slow Ramps (≈ 10 Seconds)
- Thrust Variation - 4/1 in Flight and 20/1 Between Flights
- Long Life (100 Hours)
- Improved System Weight and Size
- Propellant Selection

ELEMENTS OF LOW THRUST PROPULSION PROGRAM

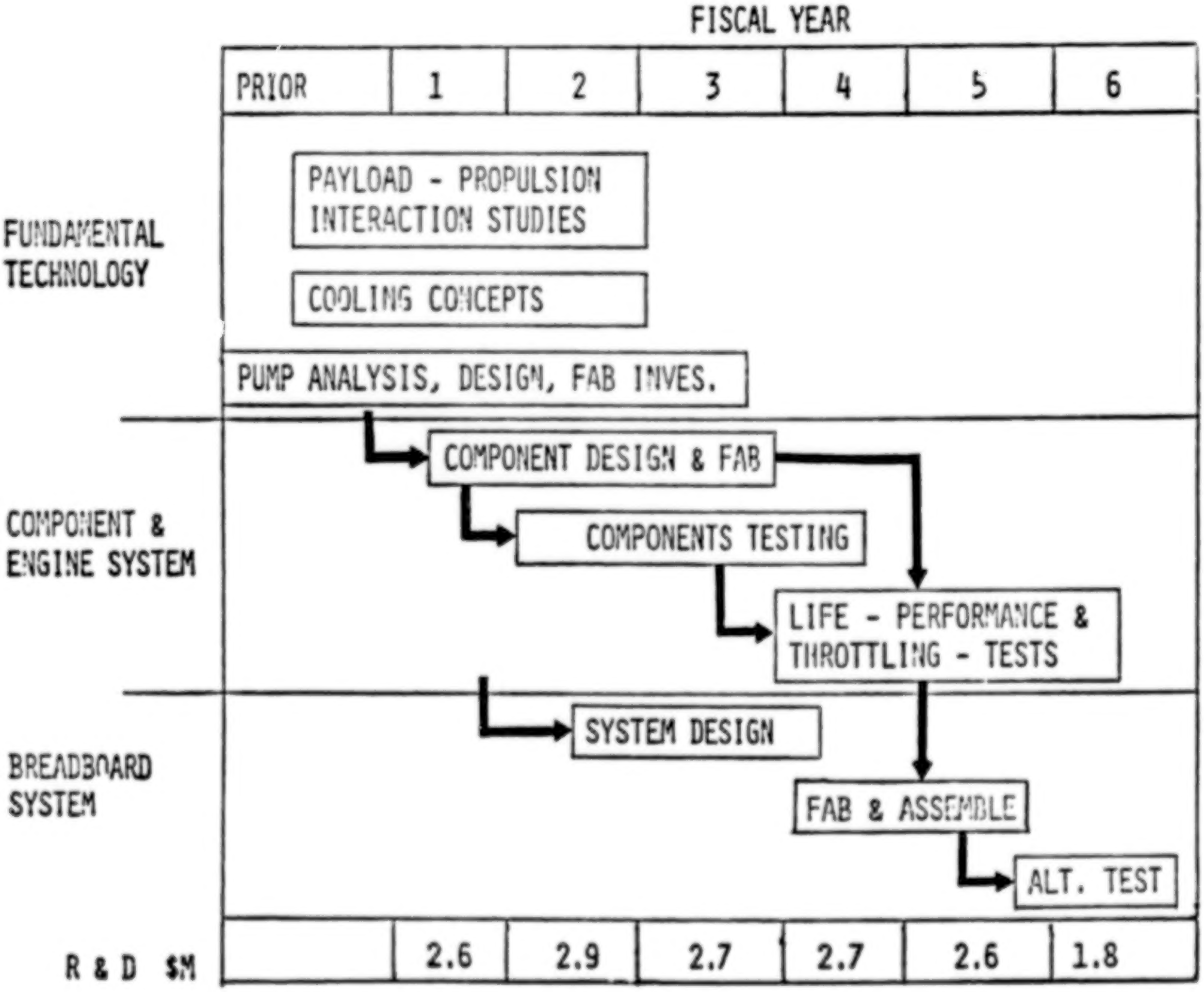
- FUNDAMENTAL TECHNOLOGIES
  - Studies to Establish Engine Requirements
  - Cooling Studies & Tests
  - Pump, Bearings, Seals, Fabrication Studies
  - Throttling Concepts
- COMPONENTS & ENGINE SYSTEMS
  - Design & Testing of Components to Demonstrate Feasibility of Critical Technology
  - Simulation Tests of Engine Systems with Most Difficult Technology
- BREADBOARD SYSTEM TEST
  - Demonstration of Technology Readiness to Achieve Life, Performance, Throttling and Multiple Start

This chart includes funding level. This is from a planning document and therefore, it shows fiscal years from when we get the increased funds that are required to accomplish the program. The first year could be the fiscal 81 or 82 program. Currently we have payload/propulsion interactions studies going on. You will hear about these later. There are cooling concepts and pump analysis studies that are being conducted, and you will also here about these later.

The next phase of the program will consist of component design, fabrication and testing in the critical technology areas. This would lead to life and performance tests to demonstrate the technology. The final phase, which I am not sure will be required, would include a complete breadboard of the system. Breadboard means not necessarily lightweight, but creation of the conditions that are needed for an engine. This would demonstrate that the technology for a complete system is available. We definitely would carry it through the design phase of the program. The final phase consists of altitude testing, because of the very large expansion ratio nozzles which have not been demonstrated to date.

That summarizes the chemical propulsion program as we see it now. The funds shown are what we think are required to do the program. This funding level is in the FY82 budget. We are planning for it. On the other hand, it is roughly double or triple the funds we have available right now for the program.

LOW THRUST PROPULSION PROGRAM


FISCAL YEAR
PRIOR
1
2
3
4
5
6
FUNDAMENTAL TECHNOLOGY
PAYLOAD - PROPULSION INTERACTION STUDIES
COOLING CONCEPTS
PUMP ANALYSIS, DESIGN, FAB INVES.
COMPONENT & ENGINE SYSTEM
COMPONENT DESIGN & FAB
COMPONENTS TESTING
LIFE - PERFORMANCE & THROTTLING - TESTS
BREADBOARD SYSTEM
SYSTEM DESIGN
FAB & ASSEMBLE
ALT. TEST
R & D $M
2.6
2.9
2.7
2.7
2.6
1.8

LSS/PROPULSION INTERACTIONS STUDIES

Omer F. Spurlock
NASA Lewis Research Center

LSS/PROPULSION INTERACTIONS STUDIES

- PROPULSION REQUIREMENTS
- PROPULSION TECHNOLOGIES
- INTERACTION ISSUES/PROBLEMS
- LSS/STATIC LOAD INTERACTION ANALYSIS
- CONCLUSIONS

Propulsion requirements for LSS missions are similar to requirements for current missions, except that demands on both primary and auxiliary propulsion may be greater for LSS missions than they are for current missions, for reasons that will be discussed later. The only propulsion requirement peculiar to LSS spacecraft is figure control, as current spacecraft are rigid or virtually so.

PROPULSION REQUIREMENTS FOR LSS MISSIONS

- ▲ PRIMARY PROPULSION
  - LAUNCH TO LOW EARTH ORBIT
  - ORBIT TRANSFER
- ▲ AUXILIARY PROPULSION
  - ORBIT TRANSFER
  - STATION KEEPING
  - FIGURE CONTROL
  - POINTING

The applicable propulsion technologies for LSS are listed on this figure.

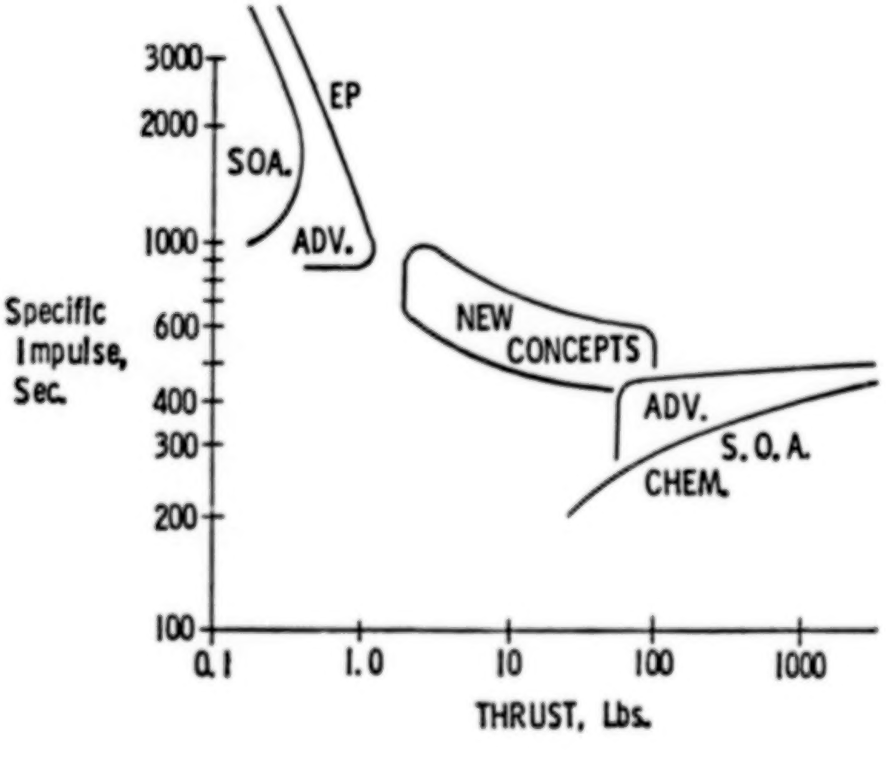
APPLICABLE PROPULSION TECHNOLOGIES

- ELECTRIC
- CHEMICAL
  - HIGH THRUST
  - LOW THRUST
- ADVANCED CONCEPTS

This figure describes the current status of low-thrust technology and the direction in which technology development is heading. Electric propulsion is characterized by low-thrust levels but high specific impulses. Improvements in the state of the art are directed toward increasing the thrust level without great sacrifice of specific impulse. Chemical propulsion, on the other hand, is characterized by relatively high thrust but low specific impulse. Technology efforts in chemical propulsion are aimed at improving the specific impulse and extending the lifetime of low-thrust propulsion systems.

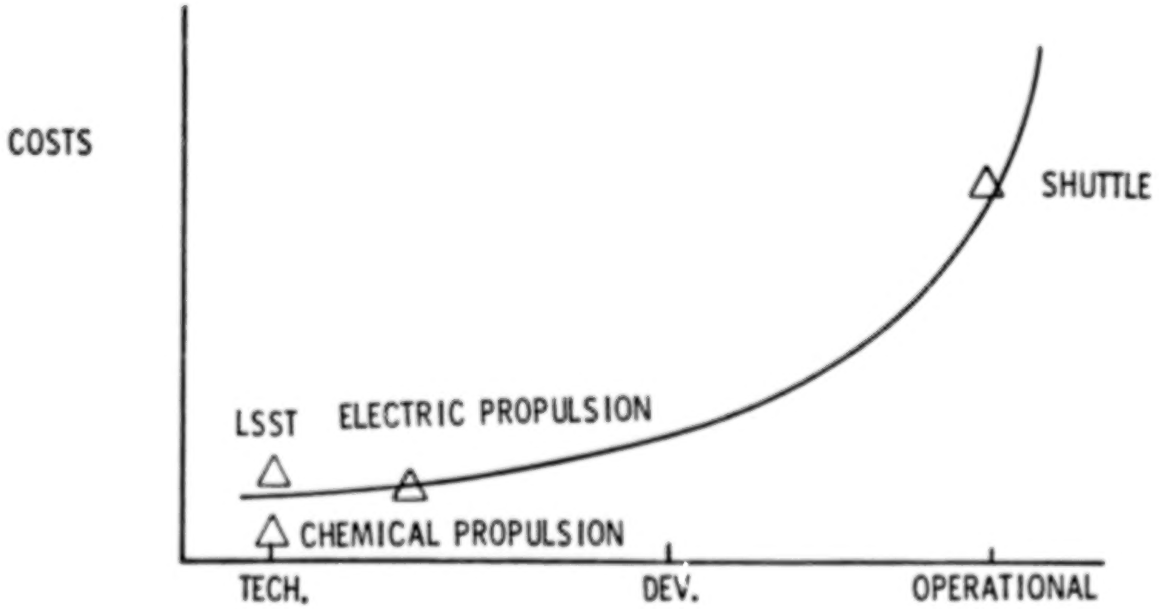
New concepts in propulsion tend to lie in the region between electric and chemical propulsion both in terms of thrust level and specific impulse.

PROPULSION TECHNOLOGY STATUS



With the exception of the Apollo program, virtually all spacecraft to this point were designed to satisfy the launch environment associated with an existing launch vehicle (usually a derivative of military development). With very minor exceptions, all compromises were of necessity on the spacecraft side of the interface. With LSS and low-thrust propulsion, we are in a new situation which offers many opportunities to optimize the propulsion/LSS system to maximize capability at minimum cost. The "cartoon" illustrates the opportunity we have. LSST and chemical propulsion are at the technology level. Electric propulsion, at least in certain respects, is moving toward the development level. Of the required components, only the Shuttle has reached the operational level where changes to specifically accommodate LSS would be prohibitively expensive. If we direct our technology efforts wisely, we can anticipate problems and grasp opportunities to maximize capability and minimize costs. Our failures will become progressively more expensive to correct as we move toward the operational stage.

COST IMPACT OF PROGRAM DECISIONS



-FAILURE TO ANTICIPATE PROBLEMS/OPPORTUNITIES MORE COSTLY AS PROJECTS MATURE

The next several charts are an attempt by LeRC to scope the LSS/propulsion interface problem from the propulsion point-of-view. Specific results have been avoided to highlight the many interactions that exist. The various areas of interaction between the propulsion system and LSS are outlined. The triangles indicate areas of interaction that are or have been investigated by LeRC or its contractors.

INTERACTION ISSUES / PROBLEMS

- STRUCTURAL EFFECTS
  - △ STATIC LOADS
  - △ DYNAMIC LOADS
  - - LAUNCH LOADS
  - △ CONTROL INTERACTIONS
  - △ THRUST DISTRIBUTION
  - △ THROTTLING ($\approx$ CONSTANT T/W)

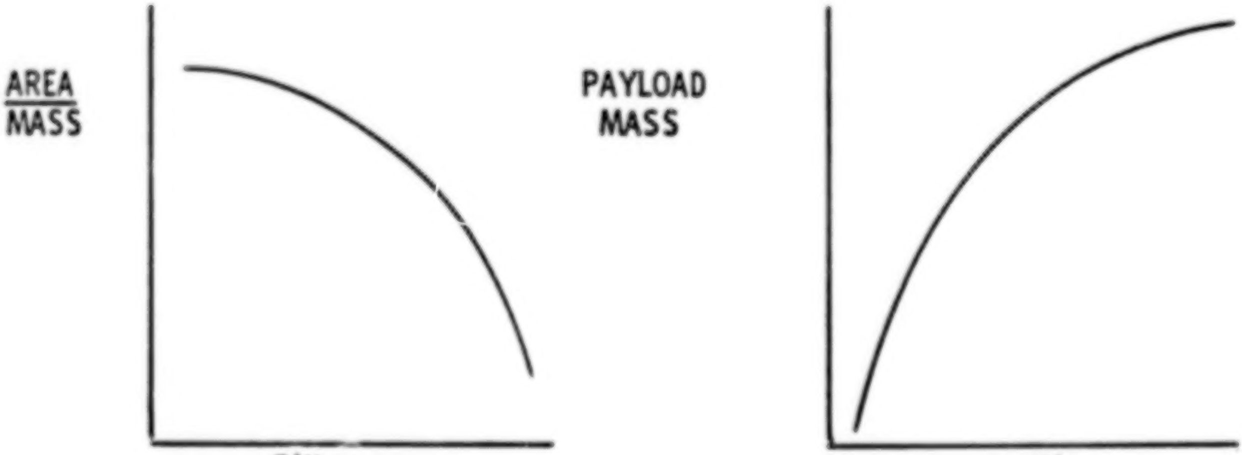
△ INDICATES ON-GOING OR COMPLETED LeRC ACTIVITY

This figure illustrates the static load/LSS interaction problem. On the left, the effect of T/W on AREA/MASS is shown, indicating that as T/W increases, the structure must be "beefed-up" to withstand the loads. On the right, the payload response to T/W is shown, indicating that over the range of interest, payload increases with T/W. By combining these data, the effect of T/W or thrust on LSS area may be derived. The results of such combinations are shown in some of the following presentations. Such data are very interesting, but recognition of the specific assumptions embedded in such data is at least as important as the data themselves. Careful consideration of a wide collection of both LSS and propulsion data will be necessary to fully appreciate our situation with regard to the static load/LSS interaction.

There are data available for this particular interaction. For other interactions we may know the abscissa and ordinates, but have little or no data. Still less defined, we may be able to intuitively recognize an interaction, but have difficulty specifying the variables. Of most concern are those interactions of these complex systems which we fail to recognize and neglect to plan for.

CHARACTERISTICS OF STATIC LOAD /LSS INTERACTION

PRIMARY PROPULSION, ORBIT RAISING



- AREA INCREASES BUT PAYLOAD DECREASES AS T/W DECREASES

This chart lists some environmental interactions. Most of these interactions are independent of the propulsion choice - electric or chemical.

ISSUES / PROBLEMS (Cont'd)

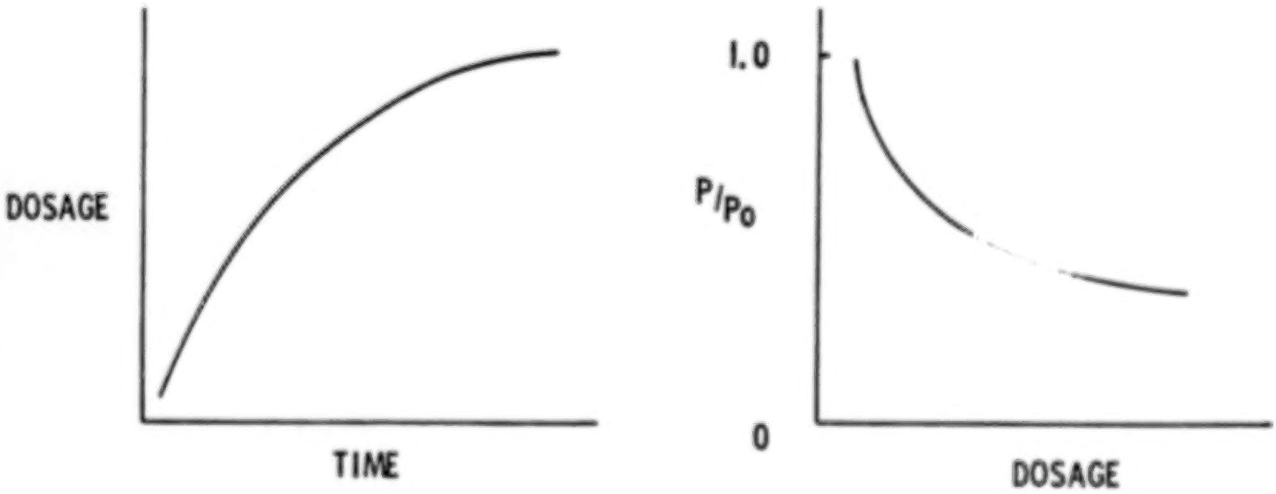
● ENVIRONMENTAL INTERACTIONS

- △ RADIATION EFFECTS
- △ LIFE & DEGRADATION
- - HEATING (PROPULSION & PAYLOAD)
- - ORIENTATION
- △ DRAG
- △ SPACECRAFT CHARGING
- △ PROPULSION EFFLUENTS

This chart illustrates one of the environmental concerns associated primarily with solar electric propulsion. As is well known, passage through the Van Allen radiation belts damages solar cells, reducing the power available for propulsion. The loss of power is a function of dosage and the susceptibility to damage of the cells. The mission design (which is spacecraft and mission dependent) affects the radiation dosage and the protection afforded the cells (by glass covers, for instance) affects the weight of the propulsion system, which in turn affects the spacecraft. If the spacecraft is supplying the power for the propulsion system, any reduction in power reduces power available for propulsion. For solar electric propulsion systems, these interactions should be considered to optimize the system.

RADIATION EFFECTS

SOLAR CELLS



- RADIATION ENVIRONMENT WILL AFFECT POWER AVAILABLE FOR PROPULSION

Control interactions between the LSS and the propulsion system promise to be some of the more difficult interactions to investigate, not only because of the modeling problems for such complex spacecraft, but also because ground testing of control systems may prove impossible. That is to say, considerable investment in space-based experimentation may be required before models can be shown to accurately represent structural characteristics.

Up to the present, no provision has been made to deorbit unclassified spacecraft when their useful lifetimes are completed. To deorbit such spacecraft, a propulsion system in working order must be available, either by a system on the spacecraft at the end of its mission or by attaching a system which has been sent to perform that task. In either case, the requirement (if real) will affect the propulsion system, propellants, structure, and/or control systems.

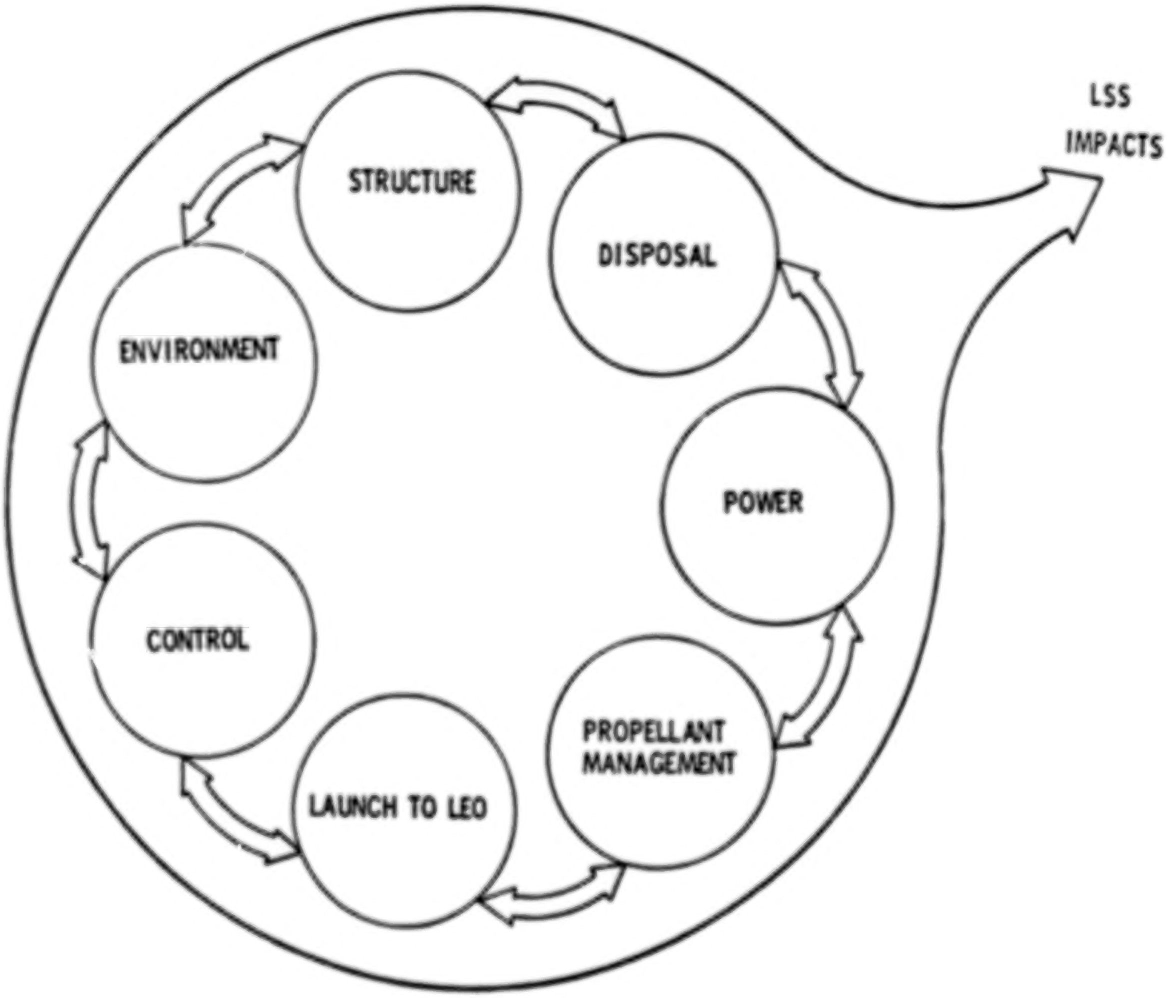
The Shuttle launch environment will also affect the spacecraft propulsion system in many ways, particularly when crew safety considerations are included in the system choice.

ISSUES / PROBLEMS (Cont'd)

- CONTROL INTERACTIONS
  - △ LARGE FLEXIBLE STRUCTURE
  - △ LIFETIME
  - △ NON-NEGLIGIBLE FORCES (GRAVITY GRADIENT & SOLAR PRESSURE)
  - RENDEZVOUS AND DOCKING REQUIREMENTS
- DISPOSAL OF DEBRIS / OBSOLETE SPACECRAFT
  - △ PROPULSION LIFETIME
  - RENDEZVOUS AND DOCKING REQUIREMENTS
- LAUNCH TO LOW EARTH ORBIT CONSTRAINTS
  - △ DENSITY
    - CENTER OF GRAVITY
    - CRADLE/BRACE PENALTIES
  - △ VOLUME LIMITATIONS

After consideration of all these interactions it becomes apparent that LSS/propulsion interactions are large, significant, interrelated, and complex. Each of the interactions affects the others in ways and to an extent not previously encountered. The results of the sum total of the interactions will greatly affect LSS spacecraft design and capability.

LSS/PROPULSION INTERACTIONS


LSS
IMPACTS
STRUCTURE
DISPOSAL
ENVIRONMENT
POWER
CONTROL
PROPELLANT MANAGEMENT
LAUNCH TO LEO

To complete our list of interactions, propellant management will affect and be affected by the interactions listed up to this point in evident ways. In turn, propellant management limitations will affect those other interactions. A similar situation exists with power interactions.

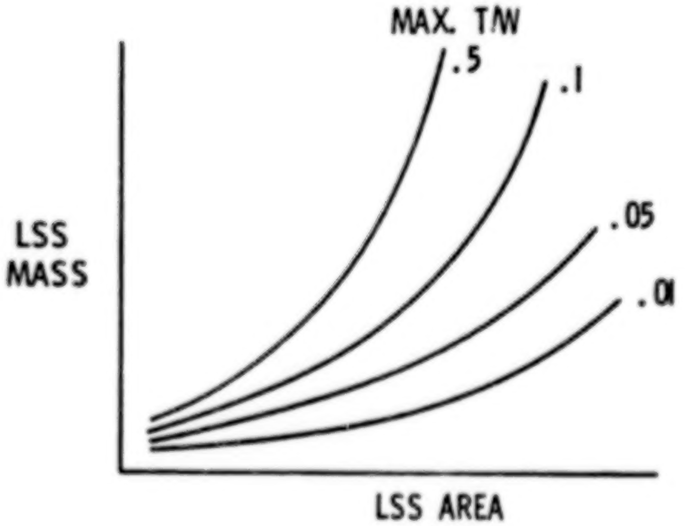
It appears clear to us that to a greater extent than was necessary (or possible) earlier, analysis of the TOTAL interaction between the spacecraft and propulsion system will be essential to providing maximum capability at minimum cost for LSS spacecraft.

ISSUES/PROBLEMS (Cont'd)

- PROPELLANT MANAGEMENT
  - △ PROPULSION CONFIGURATION
  - △ PROPELLANT CHOICE
  - △ RESTART REQUIREMENTS
- POWER INTERACTIONS
  - SPACECRAFT POWER REQUIREMENTS & AVAILABILITY
  - PROPULSION POWER REQUIREMENTS & AVAILABILITY

To return to a discussion of the investigation of the static load/LSS interaction. The next four viewgraphs are a description of on-going in-house analytical activities in this area. The information on figure 14 is characteristic of the type of data needed to describe the sensitivity of LSS mass to T/W ratio. There are limited data of this sort available and they will vary significantly for different LSS concepts. Before an adequate determination can be made of the proper thrust level for a low-thrust chemical propulsion system, data of this type representative of the spectrum of large space structures will be needed.

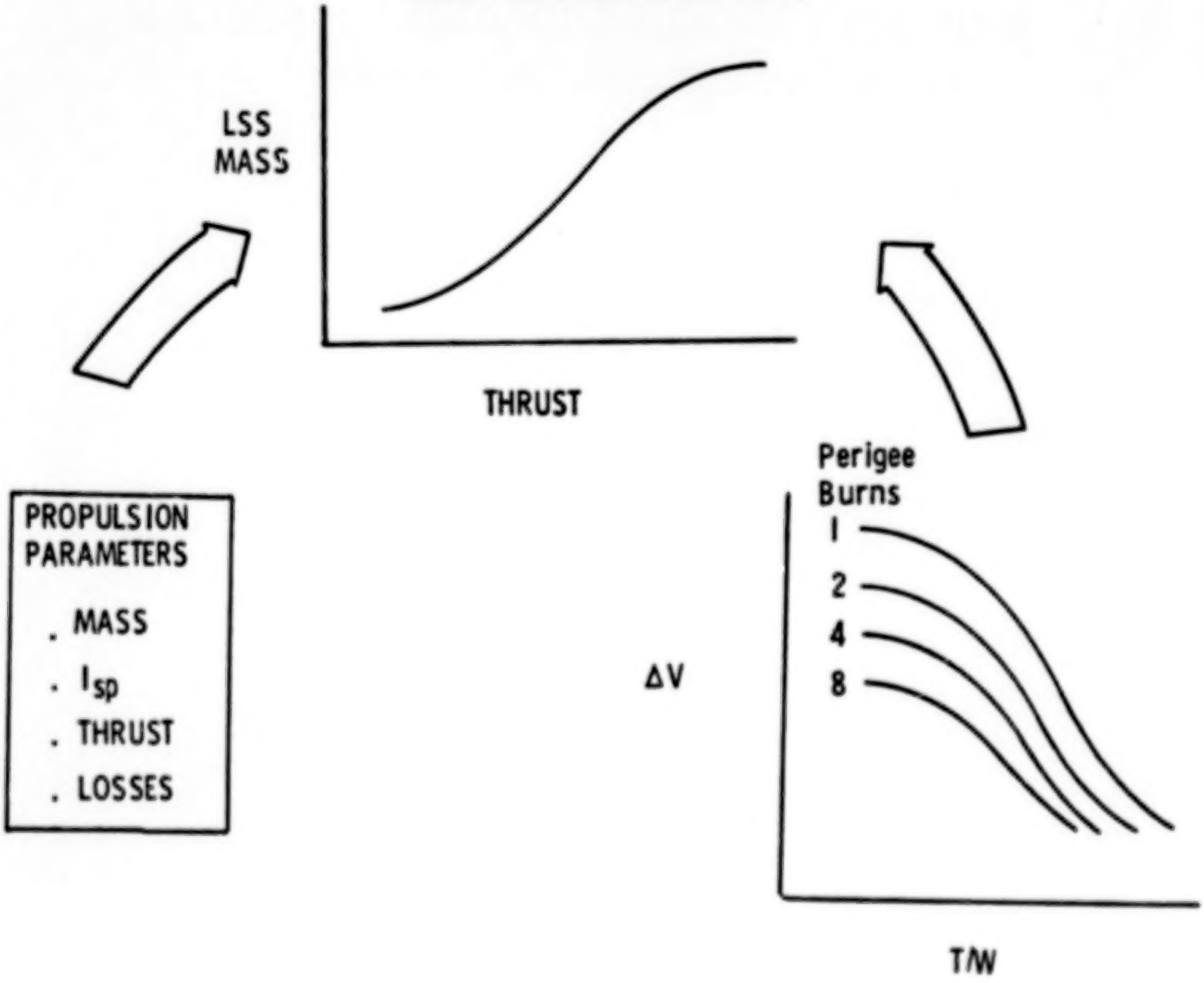
PRELIMINARY INVESTIGATION OF STATIC LOAD / LSS INTERACTION
PRIMARY PROPULSION, ORBIT TRANSFER



- LIMITED DATA AVAILABLE
- WILL VARY SIGNIFICANTLY FOR DIFFERENT CONCEPTS

On the propulsion side, performance data as a function of T/W ratio are required. These data are dependent on propulsion parameters (as shown) and on trajectory assumptions (ΔV). The ΔV data available for the thrust-to-weight levels characteristic of low-thrust propulsion systems are not minimum. The trajectories are not optimum. LeRC is sponsoring a grant with Dr. John Breakwell of Stanford to investigate this problem.

PRELIMINARY INVESTIGATION OF STATIC LOAD/LSS INTERACTION



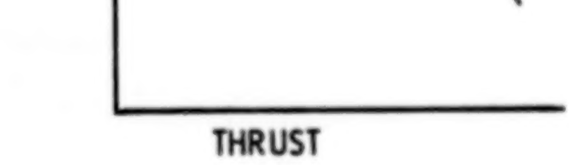
By combining information from figures 14 and 15, LSS area as a function of thrust level may be obtained. We are interested in obtaining a spectrum of such data in order to span the region of interest and understand the relationship between propulsion system thrust level and LSS area.

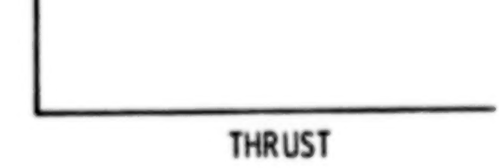
Also of interest is the cost per unit area as a function of thrust. Data of this sort are necessarily less precise than area/performance calculations, but may be helpful in understanding if influential factors involved in costs are understood.

PRELIMINARY INVESTIGATION OF STATIC LOAD/LSS INTERACTION

PRIMARY PROPULSION, ORBIT TRANSFER

- FIXED INITIAL MASS



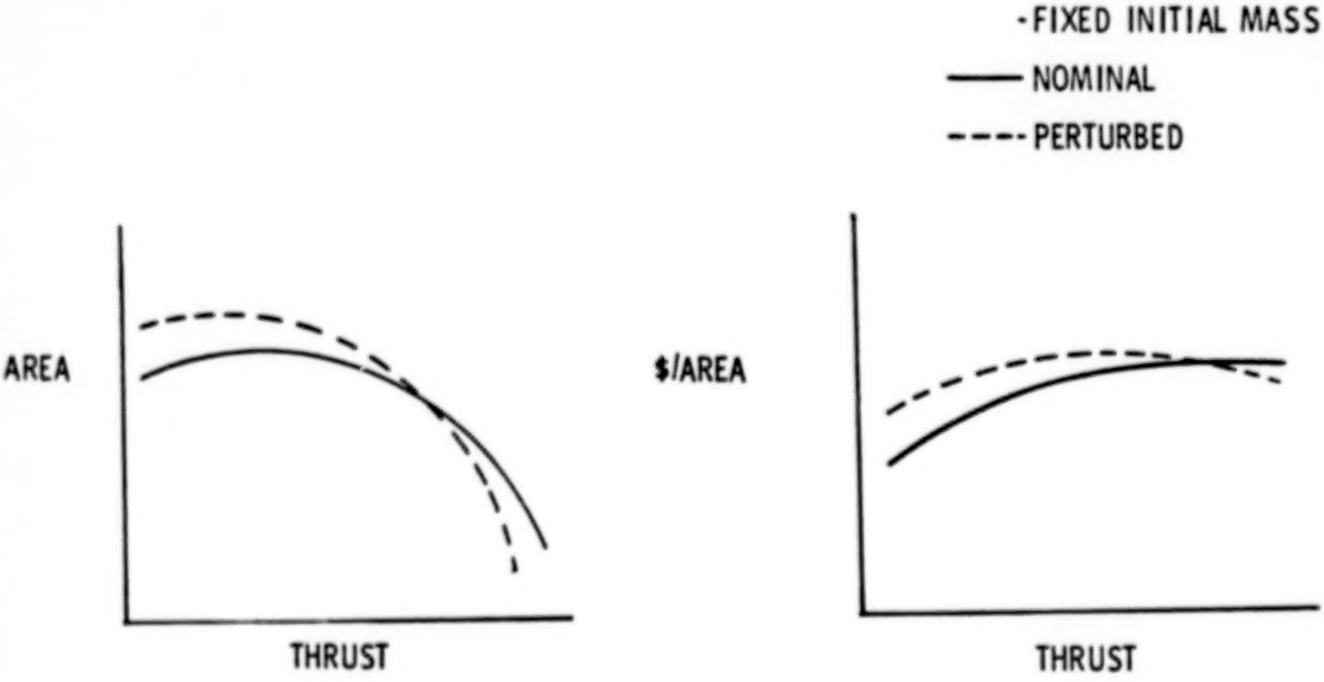


In planning technology direction it will be helpful to perform perturbation or sensitivity studies in order to understand the impact of altering propulsion or trajectory parameters and to evaluate the influence of such parameters on capability or cost.

PRELIMINARY INVESTIGATION OF STATIC LOAD/LSS INTERACTION

PRIMARY PROPULSION, ORBIT TRANSFER

PERTURBATION STUDIES



- PERTURBATION STUDIES MAY BE USEFUL TO EXAMINE TECHNOLOGY OPTIONS

We have identified many interactions between the propulsion system and the LSS payload. Further, we have observed that the interactions are not independent and must be evaluated together to accurately assess the total interaction of the propulsion system and the LSS payload. LeRC is investigating some of these interactions either in-house or by contracted effort.

LeRC is also convinced that because of the intensity of the interactions between the propulsion system and the LSS payload, careful collaboration between the payload and propulsion technology efforts will be required to avoid misdirection and exploit unique opportunities.

CONCLUSIONS

- MANY INTERACTIONS IN LSS/PROPULSION INTERFACE
- LeRC INVESTIGATING SOME OF THEM
- STATIC LOAD/LSS INTERACTION DISCUSSED IN SOME DETAIL
- CHANCE TO AVOID MISTAKES AND TAKE ADVANTAGE OF OPPORTUNITIES.

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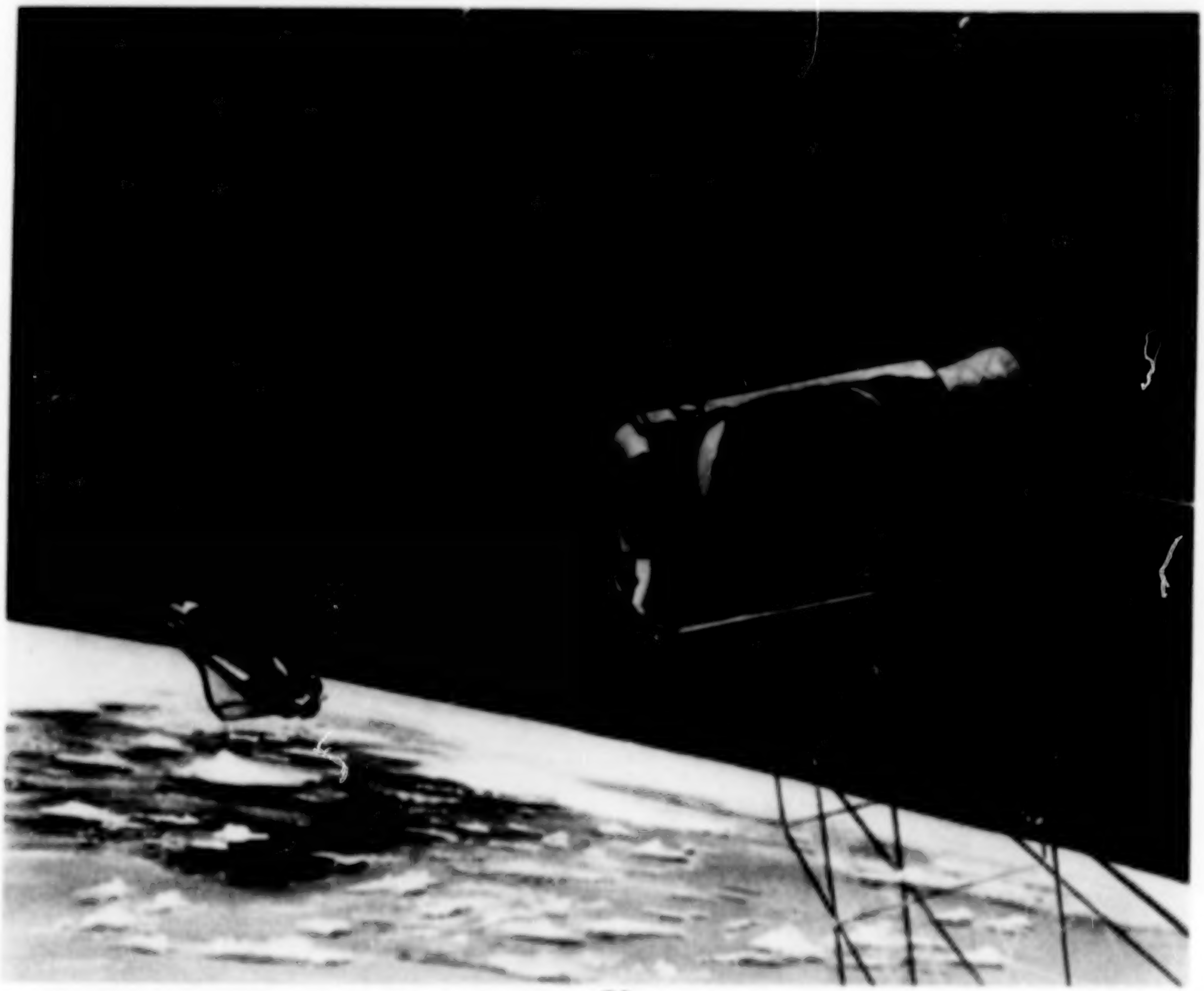
DOD LOW-THRUST MISSION STUDIES

William E. Pipes
Martin Marietta Corporation

Advanced Low Thrust Propulsion System

The Space Transportation System (STS) will be the principal means of launching USAF spacecraft beginning in the 1980's. Since it is manned and reusable it provides new opportunities for unique approaches for cost effective utilization of its capabilities. The STS also places additional requirements and constraints on advanced spacecraft deployment systems that did not previously exist for expendable launch vehicles. To fully utilize these new capabilities designers must be prepared by having cost-effective technologies available. Martin Marietta Corporation under contract to the Air Force Rocket Propulsion Laboratory (F04611-79-C-0032) performed a study to identify advanced propulsion technology that would provide flexibility, performance, and economic benefits to future Air Force missions.

The figure shown is an artist concept of an advanced low thrust propulsion system delivering a Large Space System from the Shuttle orbit to high earth orbit. This LO_2/LH_2 stage with a torus LO_2 tank and 500 lbf pump fed engine is high on the list of propulsion technology.



Study Ground Rules and Assumptions

The study ground rules and assumptions are presented here. Emphasis was placed on the military requirements for space missions planned from 1985 to the year 2000. NASA missions that complemented the DOD missions were also considered. In most cases all the Non-DOD (NASA, commercial & foreign) missions complement DOD with the exception of planetary missions. Therefore all planetary missions were excluded.

All of the missions were assumed to operate out of the Shuttle with performance and constraints defined in JSC 07700, "Space Shuttle Systems Payload Accommodations". All spacecraft deployment performance requirements are deltas from the standard Shuttle circular orbit of 160 nautical miles. By statement of work advanced STS capability such as the Advanced Military Space Flight Capability or Heavy Lift Launch Vehicle (HLLV) were not evaluated.

Study Ground Rules and Assumptions

Emphasis on Military Requirements 1985 to the Year 2000

Consider NASA Planning That Complements DOD Geocentric

STS Baseline Capability JSC 07700

- ETR; 65,000 lb, 160 n mi Circular at 28.5 deg
- WTR; 32,000 lb, 160 n mi Circular at 98 deg

Advanced STS Capability Not Considered

- Heavy Lift Launch Vehicle (HLLV)
- Advanced Military Space Flight Capability

Propulsion Concepts Considered

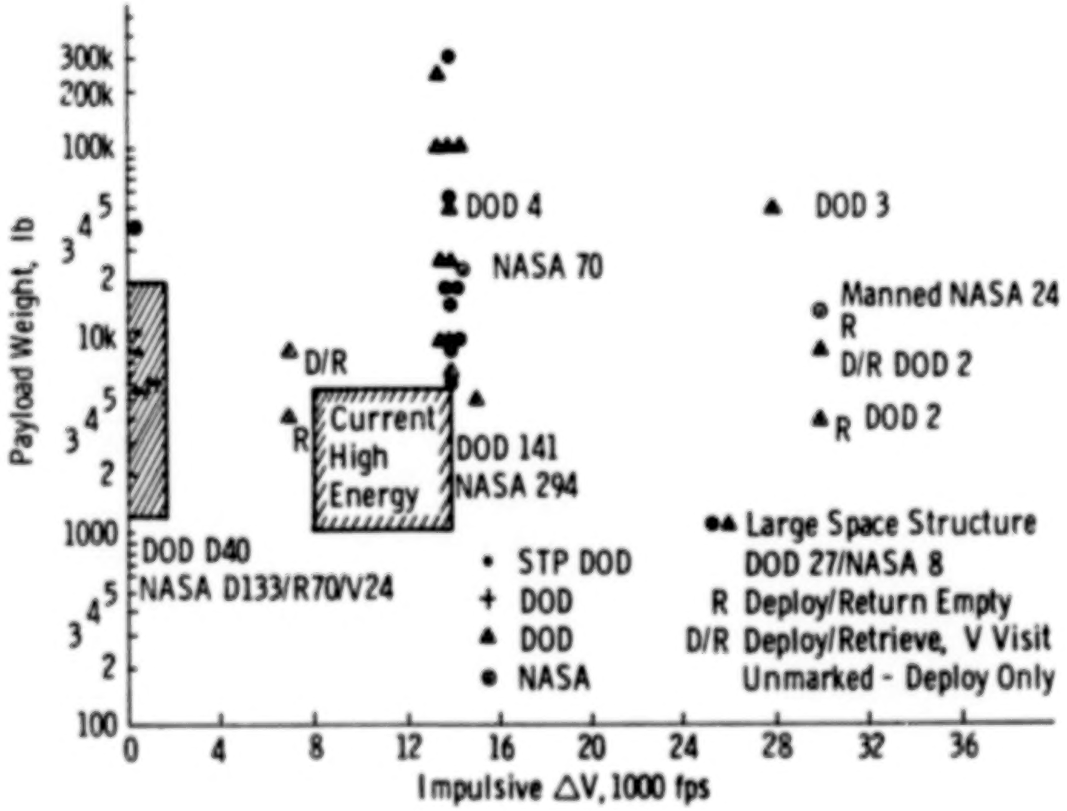
- Liquid Cryogenic and Storable (SOA and ASOA)
- Electric (SOA and ASOA)
- Solid (SOA)
- Combinations

Total Mission Catalog

The results of the Phase I mission characterization are presented here. The mission model contains low energy missions, high energy missions, and future missions which include large space systems. The quantity of missions are indicated in each area and is separated between DOD and NASA which includes commercial and foreign. As can be seen some missions are very large in weight such as the Solar Power Satellite while others require large amounts of delta velocity such as the manned mission to geosynchronous. The low energy NASA missions include deploy (D), retrieve (R), and visit (V).

The Large Space Systems (LSS) are indicated by the solid triangles and circles for DOD and NASA respectively.

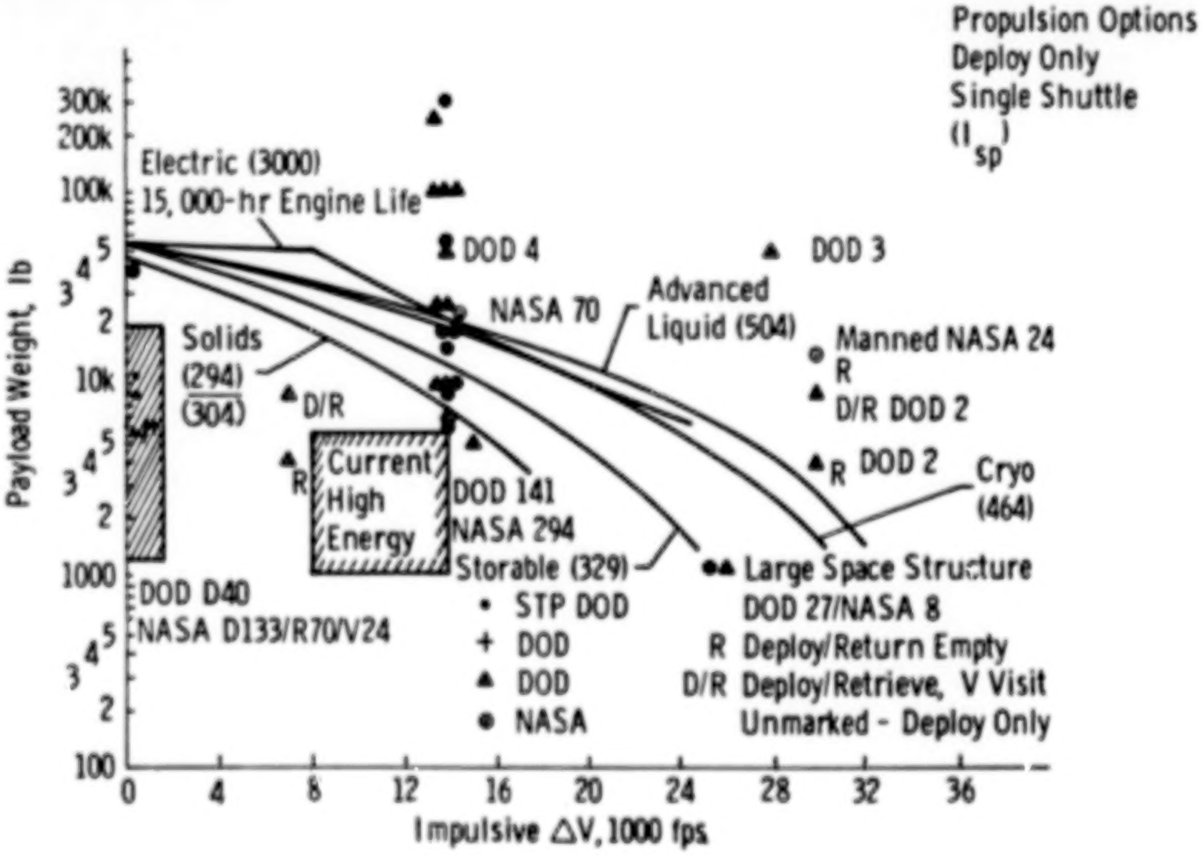
Total Mission Catalog



Total Mission Catalog

To capture these missions different deployment techniques were evaluated to determine single shuttle capability as well as multiple shuttle capability using multiple spacecraft systems. This figure compares the performance capability of the different propulsion systems. The figure includes both state-of-the-art technology and advanced technology such as the advanced liquid with 504 seconds specific impulse representing the upper limit for chemical propulsion ($LF_2/LH2$), excluding the use of metal additives which can increase the performance an additional 40 seconds. As can be seen in the figure there are Large Space Systems that cannot be captured or satisfied by a single shuttle launch.

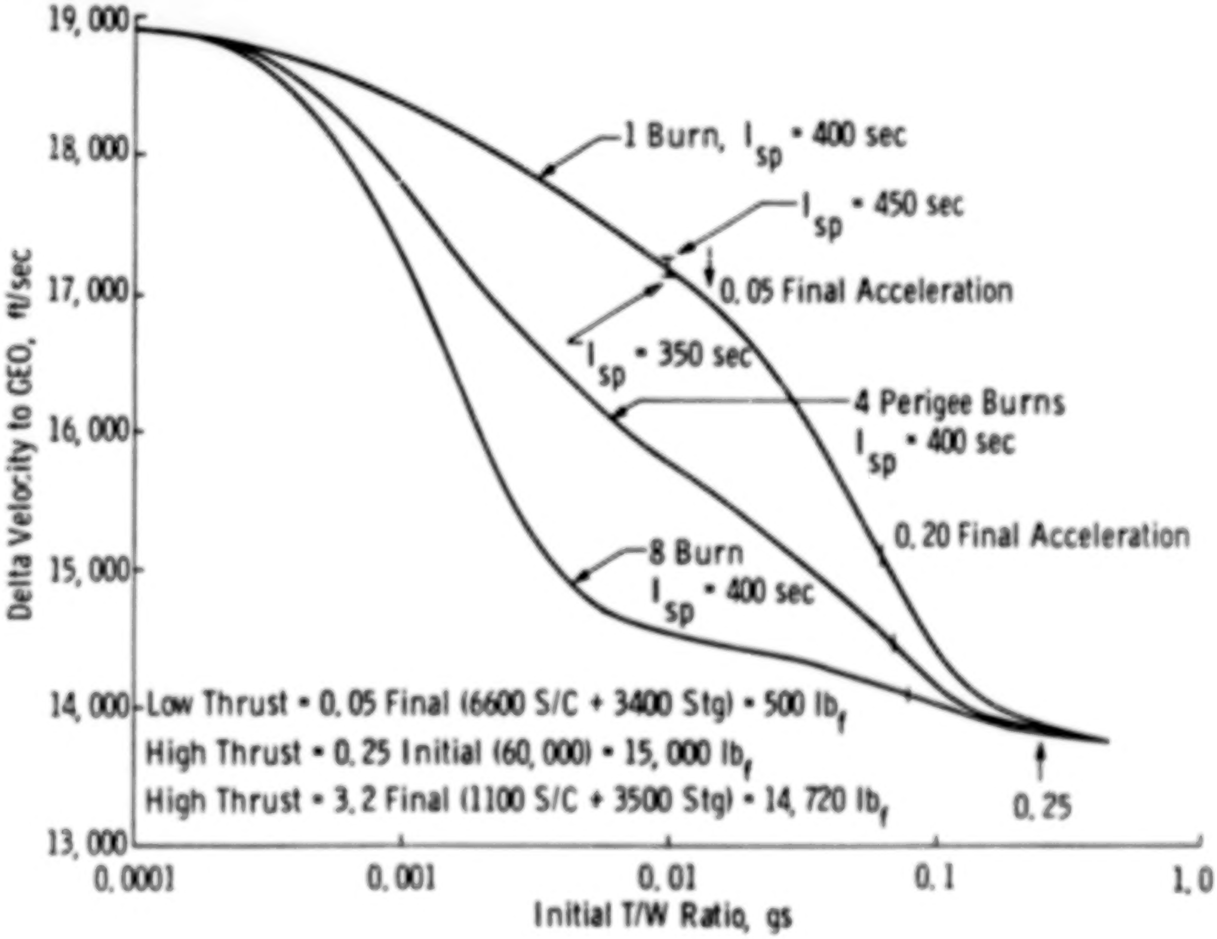
Total Mission Catalog



Acceleration and ISP Effects on Delta Velocity

Large Space Systems deployed in low earth orbit and transferred to higher orbits require low thrust to keep from exceeding their structural capability. The impact of low thrust to weight on delta velocity required is presented here. As thrust decreases to meet the LSS g-level requirements (0.05 gs) the delta velocity required to geosynchronous orbit increases due to burn inefficiencies. One way to increase performance or reduce the delta velocity required is by multiple perigee burns. The three curves are for 1, 4, and 8 perigee burns at an Isp of 400 seconds. If initial thrust to weight is at or above 0.25 g's the effects of low thrust are negligible. Using this initial point and the final burn out g-level of 3.2 for non-LSS spacecraft results in a thrust level of approximately 15,000 lb_f whereas the g-level for LSS spacecraft requires a thrust level of approximately 500 lb_f.

Acceleration and ISP Effects on Delta Velocity



Liquid Chemical Propulsion Vehicles for LSS

Presented here is a summary of the Large Space Systems requirements and the resulting vehicle requirements. With the exception of two DOD missions the spacecraft descriptions were very general with regard to orbiter packaging. The spacecraft were defined as simply one or more shuttle orbiters full. A total of 35 spacecraft were identified of which 27 are DOD. The stages were sized for the stage plus Airborne Support Equipment (ASE) and spacecraft delivery capability to equal 65,000 lbs. Mission durations were defined as a minimum of 8 days to a maximum of 60 days. The minimum value was established as approximately 7 days in shuttle orbit for spacecraft deployment and checkout and approximately 1 day (31 hrs for 8 perigee burns) for transfer to geosynchronous orbit. The 60 days was based on the requirement to assemble stages in low earth orbit to satisfy the impulse required for the larger LSS missions.

Six vehicle configurations were selected to compare the relative economic benefits of storable propellants and cryogenic propellants including an advanced combination, throttleable engine, tripropellant, and a minimum length cryogenic stage with torus LO_2 tank.

A mission capture analysis was performed for each candidate configuration with the results shown here. As indicated the lowest capture results from the advanced propellant candidate. However, the difference is small compared to the three LO_2/LH_2 concepts. The storable and tripropellant capture results are much higher due to the lower performance.

Liquid Chemical Propulsion Vehicles for Large Space Systems

Mission Description

- Spacecraft Weight Range • 6,000 to 300,000 lbm
- g-Level • 0.05 to 1.0
- All S/C Fill Orbiter Bay Except for 2 DOD
  - DOD 8 Missions/27 S/C
  - NASA 8 Missions/8 S/C

Vehicle Requirements

- Low Thrust (500 lbf)
- Spacecraft + Stage + ASE • 65,000 lbm
- 14 ft Dia x 34 ft Length (Max)
- Mission Duration (8 Days to 60 Days)
- 9 Burns Total (Max), ΔV • 14,600 ft/s

Six Concepts Identified

		Shuttle Flights	
	Length, ft	DOD	NASA
- Baseline (N_2O_4/MMH)	15.1	177	56
- Tripropellant ($CLF_5/N_2H_4/LH_2$)	25.4	156	52
- Max Perf (LO_2/LH_2)	22.7	134	47
- Max Perf (LF_2/LH_2)	18.4	132	45
- Throttleable (LO_2/LH_2)	22.3	134	47
- Minimum Length (LO_2/LH_2) - TORUS	17.0	134	47

Mission Capture Ground Rules

The mission capture ground rules used for the study are shown in the accompanying table.

Mission Capture Ground Rules

No DOD and NASA Mixing

For Grouping, the Payload Must Fly in the Same Year

Launch Site Must Be the Same

Available Shuttle Length 60 - 4 ft = 56 ft

Maximum Diameter = 14 ft

Payload Adapter Length 2 ft

Payload Adapter Weight 10% of Payload (Maximum of 1000 lb)

Grouped Payloads Require Diameter Spacing of 1 ft

Single Shuttle Flights

Reusable - Expend Only When Required for Delivery

Stage Dry Weight Contingency 10%

Flight Performance Reserve 2% (ACPS 10%)

ASE 3,000 to 5,000 lb Based on Diameter (Existing Stages Use Actuals)

LCC Analysis

To quantify the benefits of advanced technology Life Cycle Cost (LCC) was developed for each propulsion candidate based on the mission capture results. The approach to costing the propulsion candidates was to review the previous storable and cryogenic Space Tug studies and determine the major cost elements. In addition cost differences were reviewed to determine how cost would be affected by the different propulsion stage candidates. Applicable Cost Estimating Relationships (CER) were then obtained and the concepts costed based on the mission capture analysis. The costs are presented in 1980 dollars with a 95% learning curve applied.

LCC Analysis

Approach

Review Storable and Cryogenic Tug Studies

- Determine Major Cost Elements and Cost Differences
- Obtain Applicable CERs
- Cost Concepts Accordingly

Ground Rules

FY 80 $

Refurbishment Cost 30% of Unit (Reuse)

95% Learning Curve

10% Contingency Factor on All Configurations

Reliability Loss (Sensitivity)

- LCC for Resupply Includes Two Delta Missions Lost

LCC Cost Areas

The major cost elements included are: RDT&E, investment or production, operations, and shuttle launch cost. The sub-elements include avionics, structures, thermal, propulsion (tanks, engine, propellant feed, pressurization, attitude control propulsion system, and propellant), Airborne Support Equipment (ASE), systems engineering, and project management.

The costs not included are technology development, spares and logistics (which are small) facilities, and Ground Support Equipment (GSE). For facilities and GSE it was assumed that existing systems would be used or any changes would be similar for each concept. An advanced propellant loading facility was found to be small (0.1%) compared to the total LCC.

LCC Cost Areas

The Following Elements Are Included in Our Cost Analysis:

Major Elements

- RDT&E
- Investment
- Operations
- Refurbishment (Reuse)
- Shuttle Launch Cost

Subelements

- Avionics
- Structures
- Thermal
- Propulsion
    - Tanks
    - Engine
    - Propellant Feed
    - Pressurization
    - ACPS
    - Propellant
- ASE
- Systems Engineering
- Project Management
- Reliability (Sensitivity)

Cost Elements Not Included:

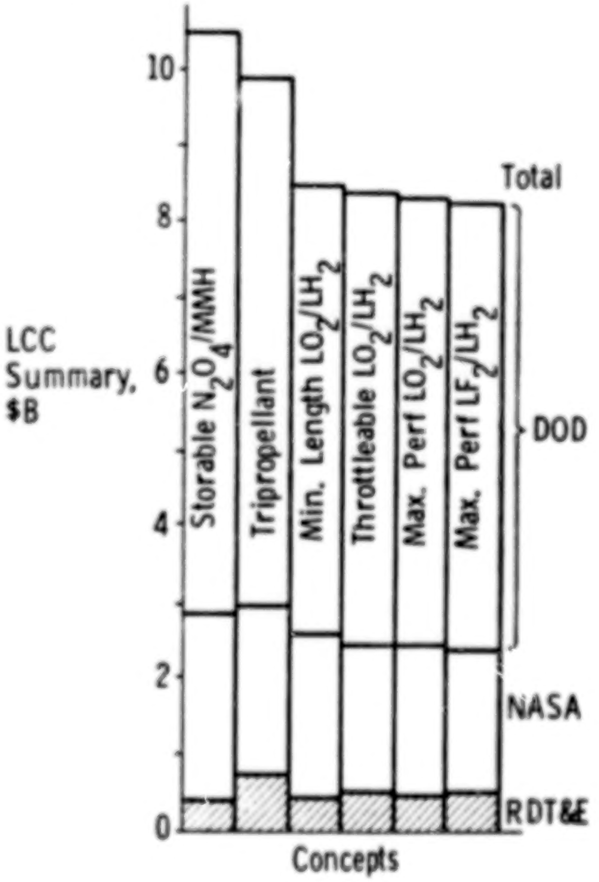
- Technology Development
- Spares
- Logistics
- Facilities (Use Existing/Changes Similar for Each Concept)
- Adv Propellant Loading Equipment (RDT&E ~ \$3.9M Unit ~ \$2.8M)
- GSE (Assumed Similar for Each Concept)

LSS Conclusions for Liquid Chemical Vehicles

Based on the mission capture analyses and LCC conclusions were reached regarding advanced technology for the LSS category of missions.

For the six stage configurations evaluated the LCC results are presented here. The results indicate that the cryogenic stage configurations are significantly lower cost than the storable and tripropellant. There are also no LCC advantages for a throttleable engine; however, interaction with the large space system due to dynamic effects may prove to be beneficial. It can also be seen that there are no LCC advantages for advanced propellants and no LCC advantage or penalty for the short torus LO_2 tank stage. This in part is due to the LSS mission definitions which in all but two DOD cases the spacecraft filled the Orbiter independent of the stage. However, from other studies performed by Martin Marietta as well as other mission categories in this study the importance of length is recognized. It is also important that the conclusions for DOD and NASA missions are the same.

LSS Conclusions for Liquid Chemical Vehicles



- Low Thrust and High Performance Required
- LCC of Cryogenic Stages Lower Than Storable Combinations
- LCC of All Cryogenic Stages Nearly the Same
- No LCC Advantage for Tripropellant or Advanced Propellants
- No LCC Advantage for Throttleable Engine
- No LCC Advantage or Penalty for Short Stage (Note LSS Mission Definitions)
- DOD Results Unchanged by NASA

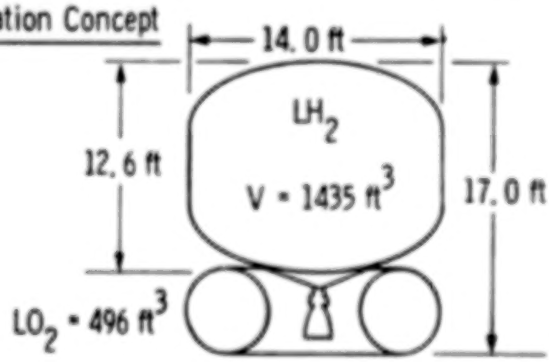
Chemical Propulsion Technology Requirements

As a result of this study the recommended chemical propulsion technology is low thrust/high performance pump fed engines combined with torus propellant tank technology. Neither of these technologies exist in a mature form and are required to meet the Large Space System requirements of the near future. The thrust level is approximately 500 lb_f and the key technology areas include small pumps, high chamber pressure, engine cooling, engine life in excess of 5 hours, and large gimbal capability. Torus tanks have not been constructed in 14 ft diameters and the propellant acquisition feed and thermal management has not been evaluated and demonstrated in these sizes or with cryogenic propellants. Summarized here are the configuration concept and key propulsion technologies. The engine performance has been updated to an Isp of 466 based on a point design provided by Aerojet Liquid Rocket Company under subcontract to Martin Marietta Corporation on the AFRPL study effort. The engine utilizes a staged combustion dual preburner engine cycle with a chamber pressure of 1000 psia.

The torus LO_2 tank was selected over other configurations based on an assessment of other tank arrangements including parallel tanks, tandem tanks, and common domes. The LO_2/LH_2 combination was also compared to LO_2/LCH_4 and found to provide nearly 1/3 more performance and for our mission model resulted in LO_2/LH_2 being the lowest life cycle cost candidate.

Chemical Propulsion Technology Requirements

Configuration Concept



Notes: Propellant LO_2/LH_2, MR = 6.0
Engine: Constant Thrust
ϵ = 400:1, Pc = 1000 psi, 96% Eff, ISP = 466 sec
Burns = 9, ΔV = 14,600 ft/s
7 Day Shuttle Orbit
Transfer Time = 31 Hours
2% Flight Performance Reserve
10% ACPS Propellant Margin
Self-Pressurization with Helium Tank for Start
Stage Weight = 44,940 lbs
Mass Fraction = 0.856
Payload Delivery = 17,060 lbs

Key Propulsion Technology

Engine Performance Demonstration
- LO_2/LH_2 Pump Fed
- Thrust = 500 lbf
- Cycles = 10
- Gimbal = ±10 deg
- Life = 5.4 hrs

Small Pumps
- Mixture Ratio Control

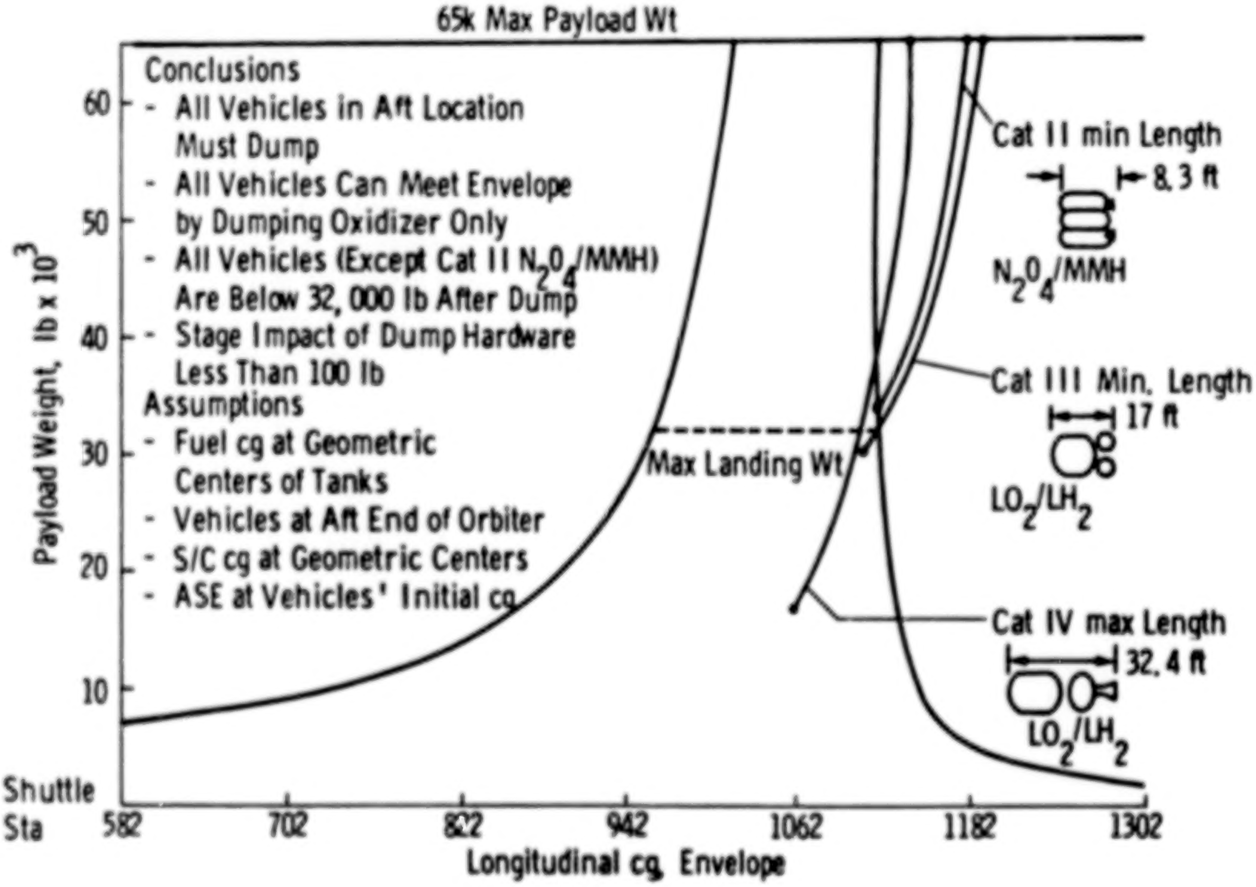
High Chamber Pressure
- Pc = 1000 psi
- ϵ = 400:1

Large Torus Tank
- 14 ft Diameter
- Weight and Manufacturing
- Propellant Management

Orbiter Payload cg Envelope During Abort

It should be noted that the large orbit transfer vehicles will require propellant dump as shown here. The most critical abort mode because of the time available is the ascent abort. This mode assumes one engine out on the Orbiter which must immediately return to the launch site since it cannot achieve orbit. The dump philosophy is to dump during powered flight above 150,000 ft. This period was selected because it provides the highest beneficial g forces, eliminates possible Orbiter ingestion, minimizes dump thrust impact on Orbiter control, minimizes the effect of center of gravity change on the Orbiter, and the propellant orientation relative to the dump outlet is the same for on-orbit dump. All vehicles must dump oxidizer to stay within the Orbiter center of gravity constraints. For this reason, parallel redundancy is required in the oxidizer system. Fuel could also be dumped; however, this imposes additional requirements on the dump pressurization system as well as requiring another set of large dump lines impacting both the stages and Orbiter. Fuel can be dumped on-orbit when time is available. For the cryogenic stages LH_2 disposal is by boil-off rather than drain; therefore, a horizontal vent is required.

Orbiter Payload cg Envelope During Abort for ASDS Vehicles



Conclusions

- All Vehicles in Aft Location Must Dump
- All Vehicles Can Meet Envelope by Dumping Oxidizer Only
- All Vehicles (Except Cat II N_2O_4/MMH) Are Below 32,000 lb After Dump
- Stage Impact of Dump Hardware Less Than 100 lb

Assumptions

- Fuel cg at Geometric Centers of Tanks
- Vehicles at Aft End of Orbiter
- S/C cg at Geometric Centers
- ASE at Vehicles' Initial cg

Electric Propulsion Vehicles for LSS

The electric propulsion analysis included five stage concepts utilizing various power options as shown here. The power options include nuclear and solar with consideration of power on the stage or spacecraft. Many of the large spacecraft require large amounts of power which can potentially be utilized by the electric propulsion system.

The stage definitions, mission capture analyses, and Life Cycle Cost (LCC) generation were prepared for comparison of the five concepts. The concepts include a baseline mercury ion with a 50 KW solar power supply, three large inert gas (Xenon and Argon) thruster systems (considered as next generation), and a magnetoplasmadynamic (MPD) system utilizing a 200 KW nuclear power source. The stage concepts were compared on the basis of how well they can deliver the required spacecraft for the LSS missions in terms of stages required, shuttle flights, and LCC.

The baseline 50 KW SEPs concept using 30 cm mercury ion thrusters was sized to meet a thrust to drag ratio of 10 for an assumed 600 ft LSS in the minimum drag orientation. This resulted in the selection of 8 BIMOD units to maximize thrust and packaging availability in the orbiter.

The number of stages and shuttle flights to capture the missions are also summarized here. The MPD has a slight increase in shuttle flights since the stage is carried up separate from the spacecraft. However, the number of stages required are approximately half that of the other concepts due to the higher performance of the MPD.

Electric Propulsion Vehicles for Large Space Systems

Mission Description

- Spacecraft Weight Range = 6,000 to 300,000 lbm
- g-Level = 0.05 to 1.0
- All S/C Fill Orbiter Bay Except for 2 DOD
  - DOD 8 Missions/27 S/C
  - NASA 8 Missions/8 S/C

Vehicle Requirements

- Solar Power Vehicles Require 1 OMS Kit
- Nuclear Power Vehicles Require 2 OMS Kits
- Spacecraft + Stage + ASE = 50,000 lbm (Except MPD)
- ΔV = 19,000 ft/s to GEO

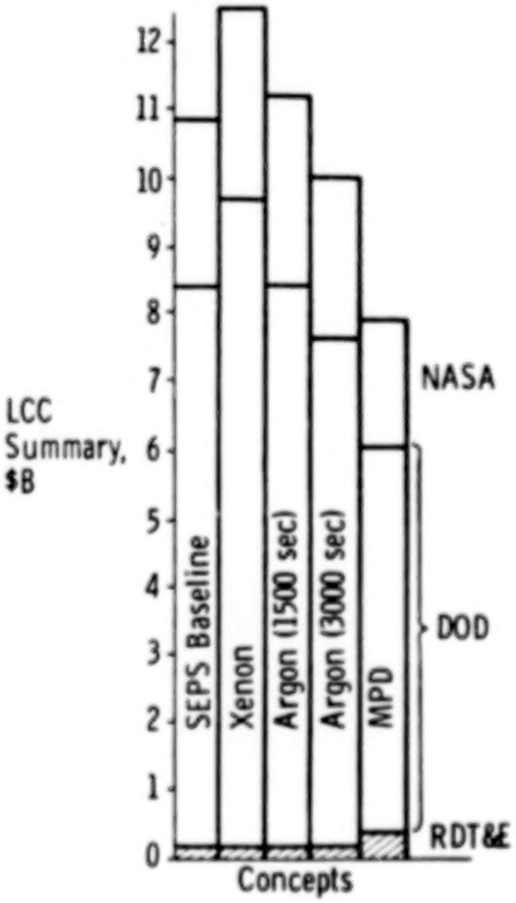
Five Concepts Identified	ISP, sec	Power, kW	Length, ft	Stages Req	Shuttle Flights
- Baseline SEPS (30 cm Hg ION)	3020	50 (Solar)	15.2	93	113
- Xenon ION Thruster	1500	50 (Solar)	15.0	106	120
- Argon ION Thruster	1500	60 (Solar)	15.0	106	120
- Argon ION Thruster	3000	76 (Solar)	15.0	98	114
- MPD	2400	200 (Nuclear)	45.0	53	122

LSS Conclusions for Electric Propulsion Vehicles

From the life cycle cost analysis the most economical electric propulsion stage is the MPD. When compared to the baseline SEPS the MPD stage is approximately 27% lower cost due to fewer stages and shorter transfer time. The Argon large inert gas thruster stage is approximately 10% lower cost. This is true for the DOD mission model and NASA mission model individually as well as the total. It is significant because it shows that the conclusions for DOD are unchanged by NASA. This is effectively a sensitivity analysis on the mission model since the DOD and NASA models differ in size, weight, frequency, and orbits.

Comparing the large inert gas thrusters to mercury ion show a slight cost advantage which in part is due to the reduction in thruster quantity. The development of these thrusters should not be on the basis of economic benefit, but on the basis of environmental impact (inert gas versus mercury) and spacecraft contamination. The use of Xenon propellant for orbit transfer is not justified due to its high cost and limited availability.

LSS Conclusions for Electric Propulsion Vehicles



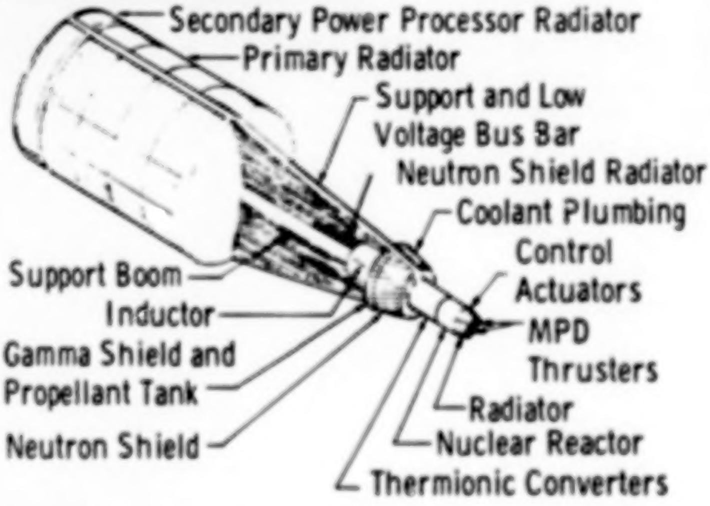
- Xenon Superior Performance Not Justified for High Traffic Orbit Transfer
  - High Cost
  - Limited Availability
- Argon Prime Candidate for Orbit Transfer Application
  - Reduced Environmental Impact
  - Reduced Spacecraft Contamination
- MPD Most Economical Electric Propulsion Concept Evaluated
- DOD Results Unchanged by NASA

Electric Propulsion Technology Requirements - MPD

The MPD key technology areas and stage configuration are summarized here. The thruster is the primary technology that should be pursued. Thrust level should be maximized by improving efficiency at the expense of weight. Related subsystems include power switching, energy storage, propellant management and thermal control, propellant flow control and isolation, and packaging of the system in Shuttle with the power supply.

Electric Propulsion Technology Requirements-MPD

Configuration Concept


Secondary Power Processor Radiator
Primary Radiator
Support and Low Voltage Bus Bar
Neutron Shield Radiator
Coolant Plumbing
Control Actuators
MPD Thrusters
Radiator
Nuclear Reactor
Thermionic Converters
Support Boom
Inductor
Gamma Shield and Propellant Tank
Neutron Shield

Length = 45 ft (Including 8 ft Dia Tank)

Key Propulsion Technology

Thruster Demonstration
- Increase Thrust
- ISP Range 1500 to 3000 sec
- Maximize Efficiency at Expense of Weight
- Life Required~15,000 hr

Related Subsystems
- Power Switching
- Energy Storage
- Propellant Management and Thermal Control
- Propellant Control and Isolation
- Packaging of Complete System
- Nuclear Power

Notes:

Power 200 kW Nuclear Power at 36 lb/kW
Efficiency = 31% MPD, Processing - 90%
Thrust - 1,067 lb_f with Extra Thruster for Redundancy, ISP = 2400 sec
ΔV = 19,000 ft/s, g-Level = 10^{-5}
Orbiter Capability = 40,000 lbs at 425 n mi

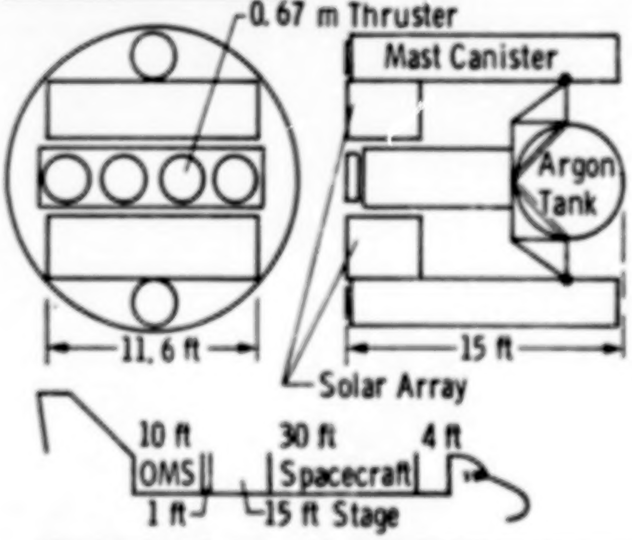
Flight Performance Reserve 2%
Transfer Time = 651 Days
Thruster Life Required = 15,625 hr
Stage Weight = 35,220 lbs
Mass Fraction = 0.71
Payload Delivery = 78,500 lbs

Electric Propulsion Technology Requirements - ION

The technology requirements for the Argon large inert gas thrusters and configuration concept are summarized here. This system has the potential to reduce contamination and environmental effects that exist with mercury. The key technology areas include propellant management, thermal control and isolation, and thruster duration testing in the larger size and its effects on the discharge chamber and cathodes both main and neutralizer.

Electric Propulsion Technology Requirements—ION

Configuration Concept



Key Propulsion Technology

Propellant Management and Thermal Control

Propellant Control and Isolation

Thruster Demonstration

- Size 67 cm
- Duration
  - Discharge Chamber
  - Cathodes, Main & Neutralizer

Notes:

Solar Power = 76 kW
Thruster Power = 17.15 kW, Eff = 42%
Thrust = 0.1088 lb_f Each, ISP = 3000 sec
Power Processing Eff = 90%
ΔV = 19,000 ft/s, g-Level = 10^{-5}
Orbiter Capability 50,000 lbs at 335 n mi

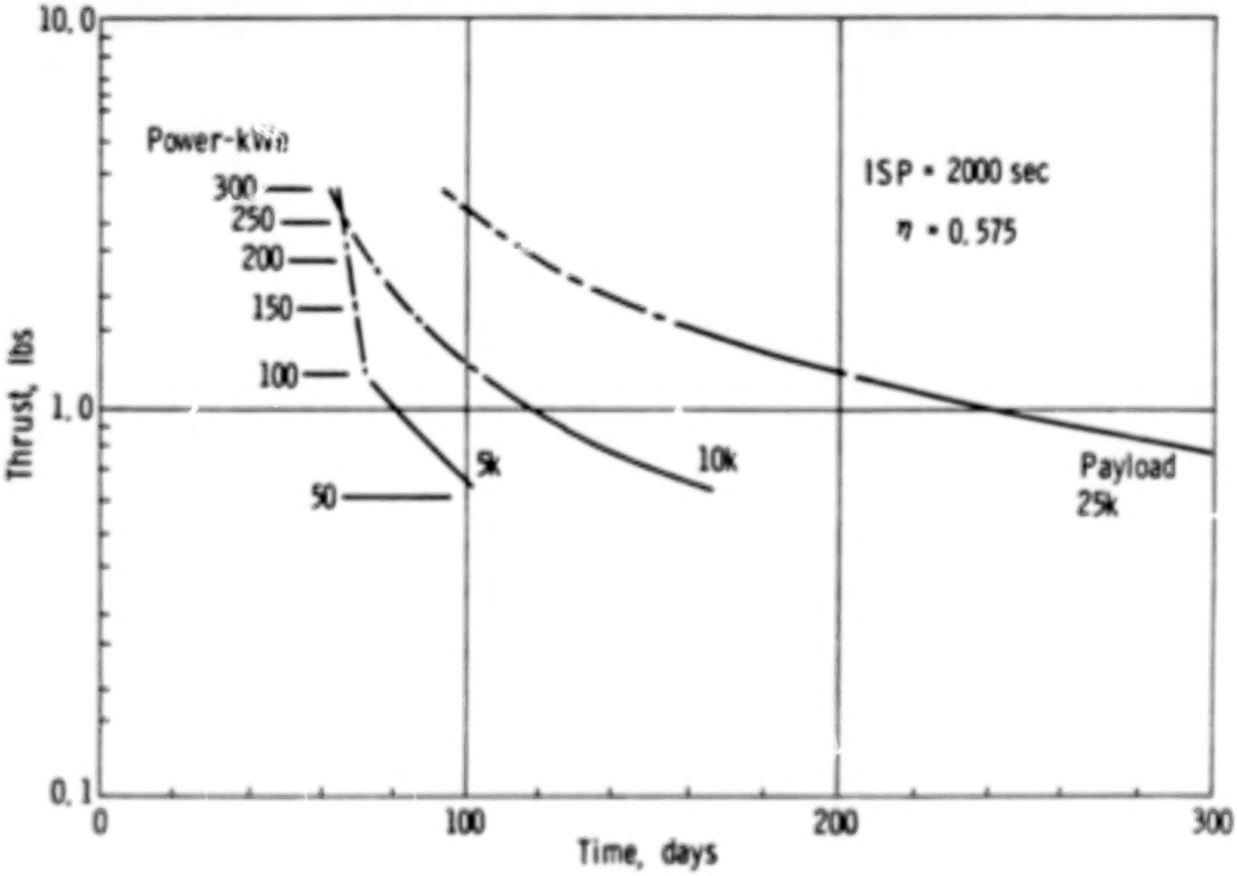
Flight Performance Reserve 2%
Transfer Time = 819 Days (Assuming 30% Solar Array Degradation Plus 5% Shadowing)
Engine Life = 14,600 hrs
Stage Weight = 12,670 lbs
Mass Fraction = 0.60
Payload Delivery = 28,600 lbs

Electric Propulsion Transfer Time from LEO to GEO

For DOD as well as NASA there are priority spacecraft and missions that must be delivered in the shortest possible time. Studies were conducted on the effect of specific impulse and efficiency on transfer time. The study results showed that a minimum transfer time of approximately 60 days to geosynchronous orbit is required to achieve any meaningful delivery capability such as 5000 lb. Shown here is an example of the study results for an Isp = 2000 sec and efficiency of 57.5%.

A significant cost factor in the LCC is the added spacecraft transfer time due to the low thrust of the electric propulsion system. To account for this it is necessary to both inflate and discount the dollar value of the spacecraft program. We followed DOD Directive 7041.3 on Economic Analysis in performing this task and found that this factor alone can be as high as 1/3 of the LCC.

Electric Propulsion LEO to GEO Transfer Time in Days



MPD Isp Optimization for 150,000 lb Spacecraft

The transfer time is determined by the system weight (stage + spacecraft) and thrust level. The higher the specific impulse the lower the stage weight or propellant weight. However, thrust decreases with increasing specific impulse by the following equation:

$$F = \frac{2n P}{g I_{sp}}$$

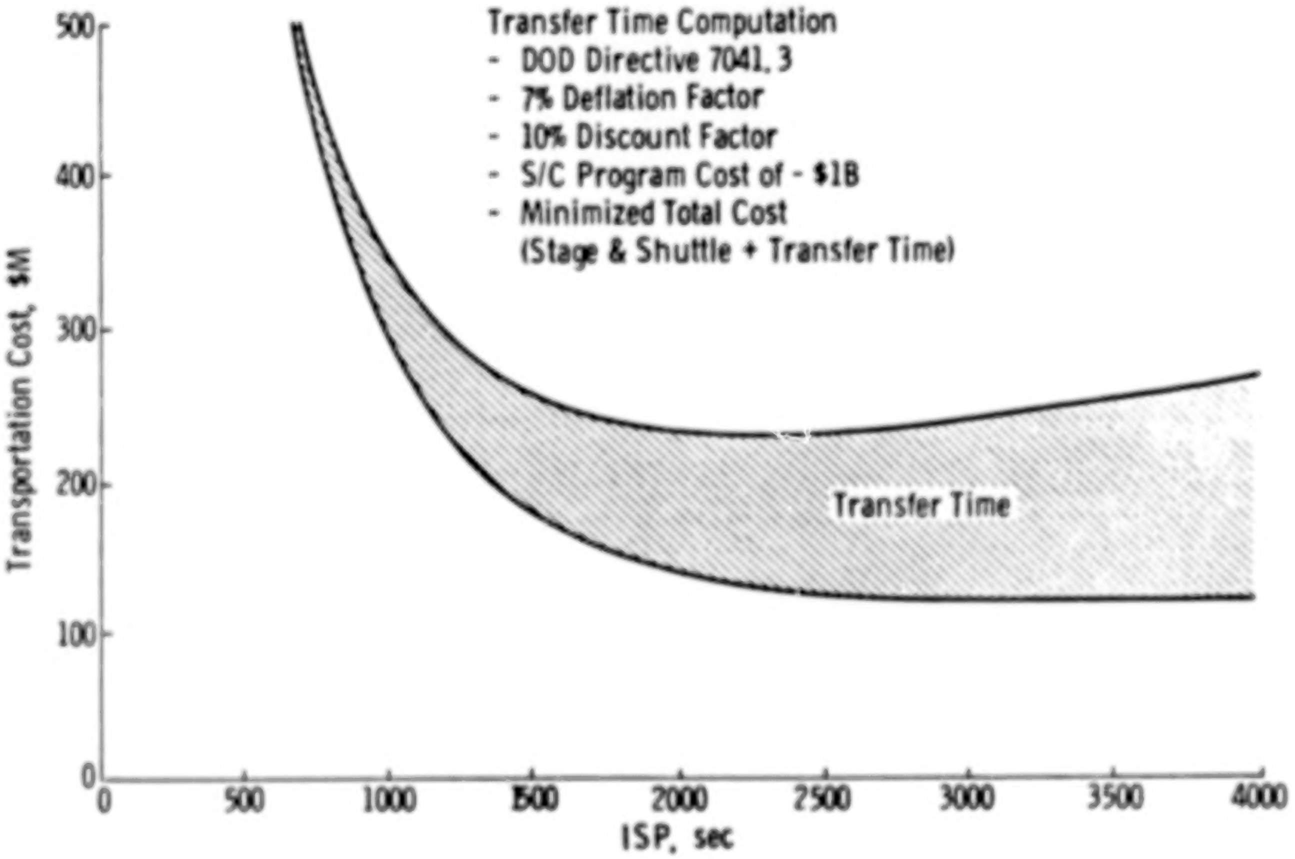
where:

F = thrust
P = electric power
n = efficiency (converting electric power to thrust)
I_{sp} = specific impulse
g = acceleration due to gravity

This decrease in thrust increases transfer time and the effect on life cycle cost.

Because of this effect specific impulse optimization studies were performed on both the large inert gas thruster (Argon) stage concept and the MPD concept. These results show the optimum specific impulse to be in the 1500 to 3000 second range as indicated here for MPD. In this range the specific impulse is high enough to reduce shuttle flights yet low enough to prevent the transfer time from negating the economic benefits of electric propulsion.

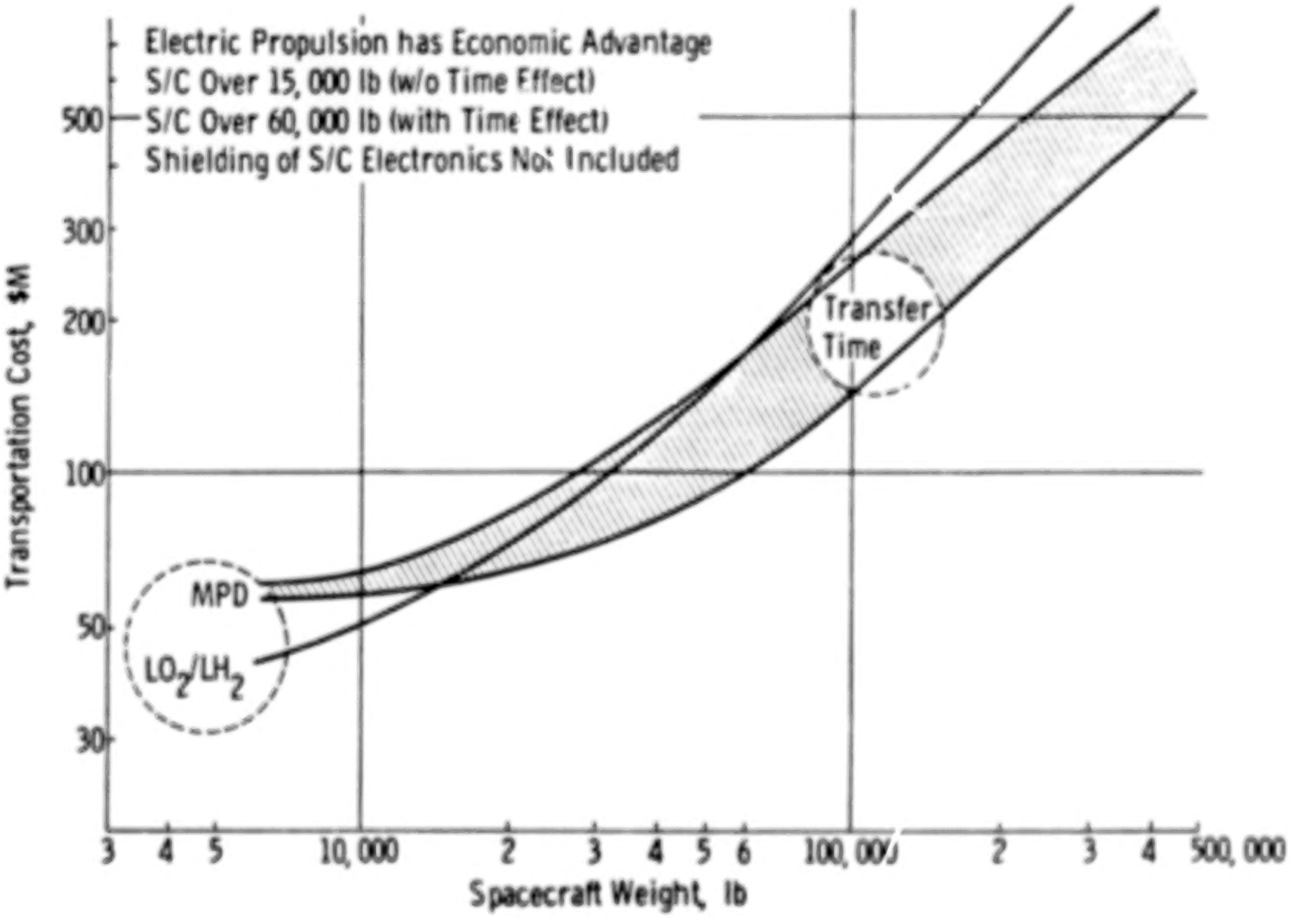
MPD ISP Optimization for 150,000 lb Spacecraft



Electric vs Chemical Propulsion

Comparing electric propulsion to chemical propulsion has shown that economic advantages can be obtained when electric propulsion is utilized on very large delivery weight systems. To better define the advantage of electric propulsion and the spacecraft weight at which the advantage begins an analysis was performed as a function of spacecraft weight independent of any mission model. The results of this analysis are shown here and represent the transportation cost for the spacecraft and stage, stage unit cost, and transfer time effect. The RDT&E for the stage is not included and the cost is for a single spacecraft at the weight indicated being delivered to GEO. The stages used for comparison are electric MPD and cryogenic LO_2/LH_2. The results show the electric propulsion stage having significant economic advantage for spacecraft greater than 60,000 lbs. If the cost of transfer time is removed the advantage occurs at a lower spacecraft weight of approximately 15,000 lbs. This saving comes primarily from reduced stages and shuttle flights due to the higher specific impulse yielding a lower weight and volume for high impulse requirements.

Electric Versus Chemical Propulsion



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LOW-THRUST VEHICLE CONCEPT STUDIES

William J. Ketchum
General Dynamics Corporation

SUMMARY

Large Space Systems (LSS) such as Geostationary Communications Platform & Space Based Radar are planned for the late 1980's and the 1990's. These are "next generation" spacecraft as large as 650 feet in size and up to 25,000 pounds in weight. Forty-seven such missions have been identified (1987-2000).

It will be advantageous to deploy and check out these expensive spacecraft in Low Earth Orbit (LEO) while still attached to the Orbiter, so any problems can be fixed, even by EVA, if necessary. The space shuttle will offer this opportunity. Once deployed and functioning, low acceleration during transfer to higher orbits (GEO) would minimize stresses on the structure, allowing larger size or lower weight spacecraft.

This report documents results of a "Low Thrust Vehicle Concept Study" conducted over a 9-month period, September 1979 - May 1980, to investigate and define new low thrust chemical (hydrogen-oxygen) propulsion systems configured specifically for low-acceleration orbit transfer of large space systems. This study for NASA/MSFC was conducted in close coordination with low-thrust engine/propulsion studies/technology efforts at LeRC and used their definitions of propulsion elements for analyses. The results of this systems/concept study are intended to help guide the propulsion technology effort already underway. This study also provides the required additional data to better compare new, low-thrust chemical propulsion systems with other propulsion approaches such as advanced electric systems.

Study results indicate that it is cost-effective and least risk to combine the low thrust OTV and stowed spacecraft in a single 65K Shuttle. Mission analysis indicates that there are 25 such missions, starting in 1987. Multiple shuttles (LSS in one, OTV in another) result in a 20% increase in LSS (SBR) diameter over single Shuttle launches.

Synthesis & optimization of the LSS characteristics and OTV capability resulted in determination of the optimum thrust-to-weight and thrust level. For the Space Based Radar with radial truss arms (center thrust application), the optimum thrust-to-weight (maximum) is 0.1, giving a thrust of 2000 lb. For the annular truss (edge-on thrust application) the structure is not as sensitive, and thrust of 1000 lb. appears optimum. For the Geoplatform, optimum T/W is .15 (3000 lb. thrust).

The effects of LSS structure material, weight distribution, and unit area density were evaluated, as were the OTV engine thrust transient and number of burns.

A constant thrust -9-burn trajectory gives better performance (and is less sensitive than constant acceleration - variable thrust) - 2-burn, and eliminates increased engine complexity (multiple low-thrust levels). Increased mission duration (3 1/4 vs 2 1/2 days total time including checkout, deployment, transfer) poses no problems for the payloads or OTV. Analysis of OTV insulation and pressurization requirements determined that propellant tank vapor residuals/pressures are little affected by engine thrust level or number of burns.

Engine thrust transient results in a dynamic factor of approximately 2. This can be reduced by using a slow, or a stepped thrust transient, but either complicates the engine, and results in little improvements in the LSS size (3%).

Distributed thrust, in addition to complicating the design of the OTV and LSS, could increase dynamic loading on the structure due to the difficulty in exact phasing of multiple thrusters.

To maximize the Orbiter payload bay volume available for the large space structure, a torus LO_2 tank is used to achieve minimum OTV length. For the 65K Shuttle, the OTV is ~18' long (allowing up to ~40' stowed payload length), having a propellant loading of 38,000 lb and a dry weight of 6000 lb.

The technology of torus tanks was investigated. A unique acquisition device was conceived that minimizes residuals no matter what the thrust offset. Only one propellant outlet is required, and no separate sumps are needed.

Several types of engines were considered; a new low-fixed thrust pump-fed engine and a low-thrust (pumped idle) mode of the OTV engine. Using 1500-lb thrust at 455 sec Isp and a 9-burn trajectory, a payload mass of ~16,000 lb can be delivered to GEO.

This study has defined an optimized low thrust OTV configured specifically for orbit transfer of large space systems. The following conclusions are made:

- Engine for an optimized low thrust stage
  - Very low thrust (< 1K) not required.
  - 1 -3K thrust range appears optimum.
  - Thrust transient not a concern.
  - Throttling not worthwhile.
  - Multiple thrusters complicate OTV/LSS design and aggravate LSS loads.

- Optimum vehicle for low acceleration missions
  - Single Shuttle launch (LSS and expendable OTV) most cost-effective and least risk.
  - Multiple Shuttles increase LSS (SBR) diameter 20%.
  - Short OTV needed which requires use of torus tank.
  - Propellant tank pressures/vapor residuals little affected by engine thrust level or number of burns.

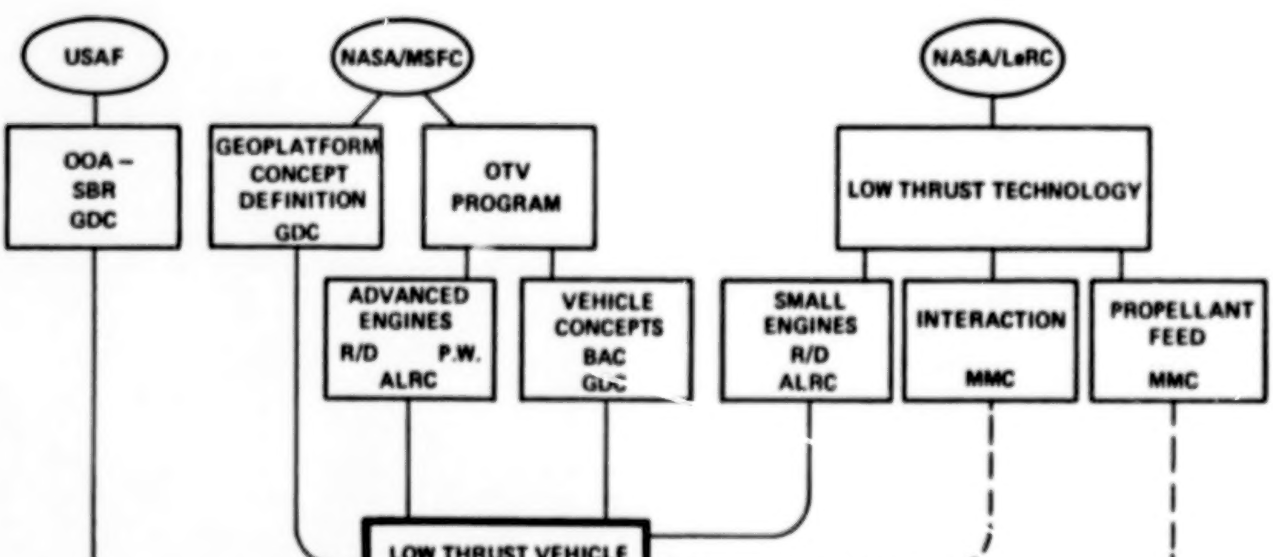
Further study is needed:

- Revise results as new mission and spacecraft data beomes available (especially as the Geoplatform design evolves).
- Re-evaluate study results as LeRC low thrust engine studies produce design concepts and cost data.
- Coordinate with OTV study (NAS8-33533 follow-on).
- Further evaluate benefits of deploying LSS at LEO vs GEO.
- Evaluate how Centaur (with idle mode) could satisfy initial requirements.
- Estimate the point at which advanced electric OTV (fast transfer/MPD) might replace low thrust chemical propulsion.

Technology development:

- Hardware R&D should be undertaken for the engines and vehicle subsystems (low thrust engine, torus tank, acquisition, insulation).

LOW THRUST CHEMICAL
PROPULSION TECHNOLOGY PROGRAM


USAF
NASA/MSFC
NASA/LeRC
OOA – SBR GDC
GEOPLATFORM CONCEPT DEFINITION GDC
OTV PROGRAM
LOW THRUST TECHNOLOGY
ADVANCED ENGINES R/D P.W. ALRC
VEHICLE CONCEPTS BAC GDC
SMALL ENGINES R/D ALRC
INTERACTION MMC
PROPELLANT FEED MMC
LOW THRUST VEHICLE CONCEPT STUDY GDC

OBJECTIVES

PROVIDE THE REQUIRED ADDITIONAL DATA TO BETTER COMPARE NEW, LOW-THRUST CHEMICAL PROPULSION SYSTEMS WITH OTHER PROPULSION APPROACHES FOR TRANSFER OF LARGE SPACE SYSTEMS.

- CHARACTERIZE MISSIONS WHICH REQUIRE OR BENEFIT FROM LOW-THRUST ORBITAL TRANSFER
- IDENTIFY, DEFINE, EVALUATE, AND COMPARE CANDIDATE LOW-THRUST LIQUID PROPULSION ORBITAL TRANSFER STAGE/VEHICLE CONCEPTS
- INVESTIGATE PAYLOAD/VEHICLE INTERACTIONS AND DESIGN IMPLICATIONS
- DETERMINE PROPULSION/SYSTEM CHARACTERISTICS HAVING THE GREATEST INFLUENCE UPON SYSTEM SUITABILITY/CAPABILITY
- IDENTIFY AND DESCRIBE PROPULSION TECHNOLOGY REQUIREMENTS

WHY DEPLOY AT LEO?

(I.E., WHY LOW THRUST?)

THE STS WILL OFFER THE FIRST OPPORTUNITY TO CONTROL, CHECK OUT, AND CORRECT THE DEPLOYMENT OF SPACECRAFT TO ENSURE OPERATIONAL READINESS BEFORE TRANSFERRING THEM TO HIGHER ORBITS.

DEPLOYMENT AT LEO CAPITALIZES ON SHUTTLE CAPABILITY AND PHILOSOPHY (MAN-ASSIST).

Mission planning (NASA and DoD) information (specifically the NASA/MSFC OTV Mission Models) was used to identify potential low-thrust missions, payload characteristics, transportation needs, and schedule requirements.

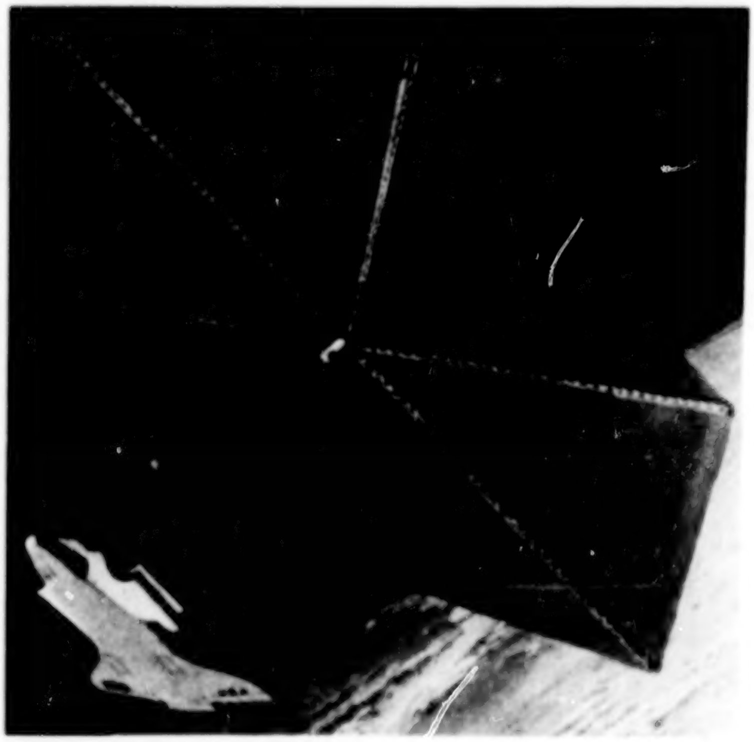
The Geoplatform Communication Antenna System, and the Space-Based Radar Antennas are the leading near-term missions. These were selected for detailed analysis. It is seen that the mission drivers are 1987 IOC; 35 ft payload; 15000 lb payload; geosynchronous mission.

A solar power array was initially considered, but was determined to be an unlikely candidate for low-thrust chemical propulsion because current concepts are designed for retraction on-orbit (protection against solar flares, etc.) and therefore it would make little sense to require transfer in the deployed condition. Future advanced (rigid-SPS, etc.) concepts will likely be self-powered (Ion or MPD engines).

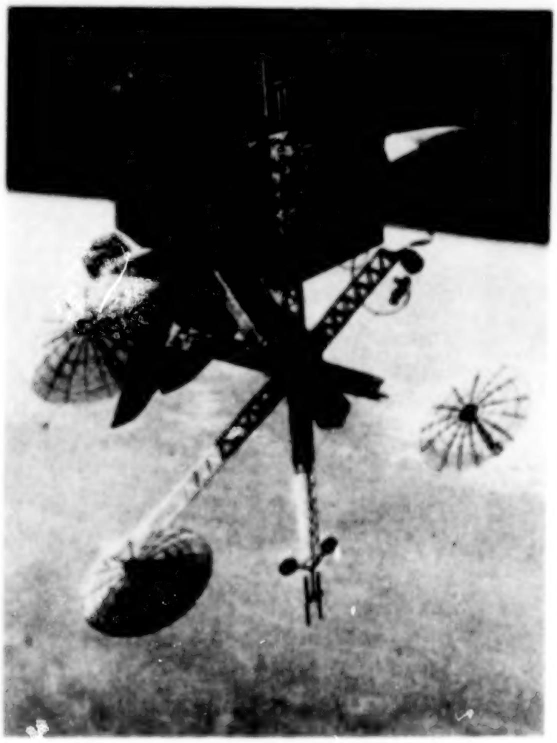
From this data, the range of requirements imposed on the OTV were determined. It is seen that for payload IOC's in the first 5 years of LSS operations (1987 - 1992) single Shuttle launches are sufficient. There are 25 such planned missions.

Starting in 1991, longer (60') and heavier (25K) payloads will require multiple Shuttle operations and use of the larger OTV being defined in a separate study (NAS8-33533).

MISSIONS/PAYLOADS



SPACE BASED RADAR



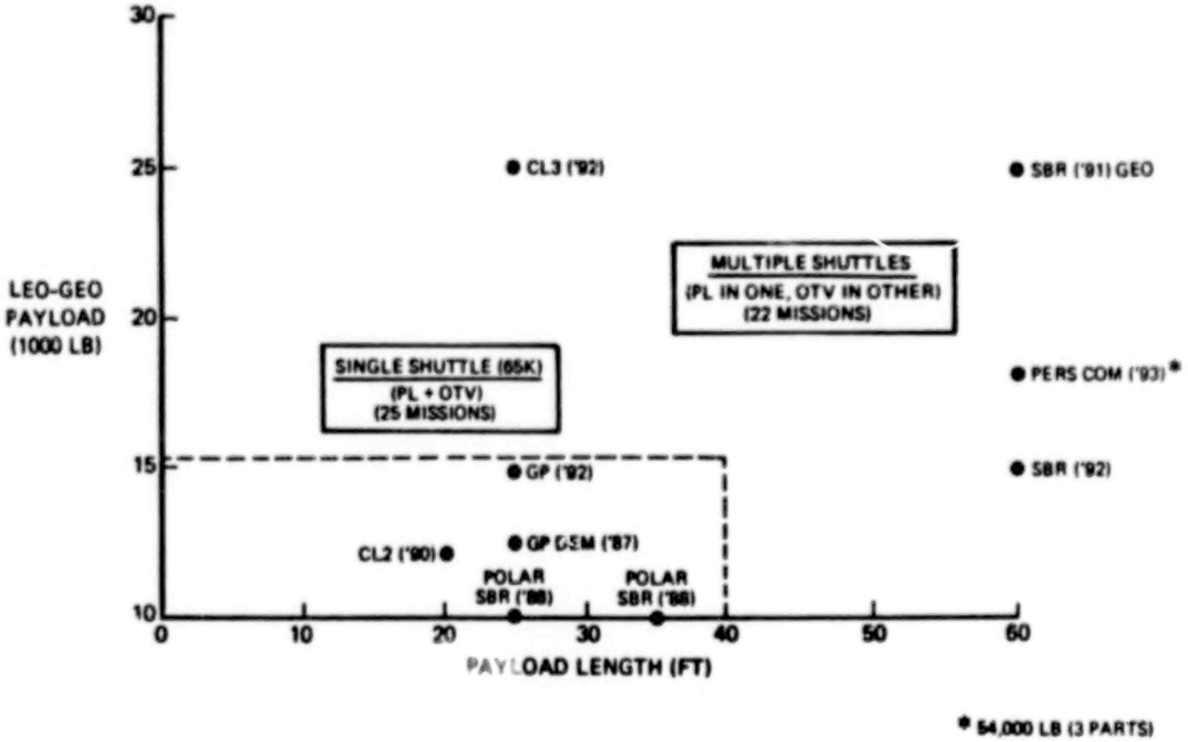
GEOPLATFORM

POTENTIAL MISSIONS/PAYLOADS FOR LOW THRUST PROPULSION

	NUMBER	IOC	
GEO-PLATFORM DEMO - 12,500 LB × 25 FT	1	1987	NOMINAL MODEL
GEO-PLATFORM - 15,000 LB × 25 FT	12	1992	
SPACE BASED RADAR			
POLAR - 10,000 LB × 25-35 FT	8	1988	
GEO - 15,000 - 25,000 LB × 60 FT	2	1991	
DOD CLASS 2 - 12,000 LB × 20 FT	4	1990	
DOD CLASS 3 - 25,000 LB × 25 FT	8	1992	
PERS COMM - 54,000 LB (3 PARTS) EACH - 18,000 LB × 60 FT	12	1993	
	47		
X-RAY TELESCOPE/GRAVITY WAVE INTERFEROMETER (SPACE FAB)		1997	MAX MODEL
SOLAR POWER DEMO (SPACE FAB)		1995	

(REF NASA MSFC 29 FEB 1980)

PAYLOAD ALLOCATION



DESIGN & OPERATIONAL CHARACTERISTICS OF SELECTED PAYLOADS

	SBR		GP	
	POLAR	GEO	EXPER	OPR
DESIGN CHARACTERISTICS				
WEIGHT (LB)	10,000	15,000- 25,000	12,500	15,000 (NOM)
STOWED LENGTH (FT)	25-35	60	25	25
OPERATIONAL CHARACTERISTICS				
MISSION	5600 N. MI. POLAR	GEO	GEO	GEO
IOC	1988	1991	1987	1992
FUNCTION	AIRCRAFT SHIP, GROUND VEHICLE SKIN TRACKING	SAME	ADVANCED COMMUNICATION AND EARTH OBSERVATION	ADVANCED COMMUNICATION AND EARTH OBSERVATIONS
LIFE	10 YR	10 YR	5 YR	16 YR (NOM)
SERVICING	NO	NO	TEST	EVERY 1-1/2 YR

⬭ IMPACTED BY OTV   REF: NASA/MSFC 29 FEB 1980

SELECTED MISSIONS ARE THE GEOPLATFORM AND SPACE BASED RADAR. DRIVING REQUIREMENTS ARE: 1987 IOC; 25-35 FT PAYLOAD LENGTH; 15,000 LB PAYLOAD WEIGHT TO GEOSYNCHRONOUS ORBIT.

GEOSTATIONARY PLATFORM PROGRAM

MISSION GOALS

- MAXIMIZE EFFICIENT USE OF AVAILABLE FREQUENCY SPECTRUM THROUGH FREQUENCY REUSE AND OTHER ADVANCED TECHNOLOGIES.
- REDUCE CONGESTION IN THE GEOSYNCHRONOUS ORBITAL ARC.
- REDUCE COSTS BY SUBSYSTEM SHARING AND "ECONOMY OF SCALE".
- USED PRIMARILY FOR COMMUNICATIONS (COMMERCIAL, NASA, AND DOD) BUT ALSO OFFERS TENANCY AND SUPPORT FOR EXPERIMENTS, ETC.

BACKGROUND

- NASA/MSFC PHASE A CONCEPTUAL DEFINITION CONTINUING BY GDC WITH COMSAT, COORDINATED WITH COMMERCIAL INTERESTS.

CONCEPTS

- RANGE FROM VERY LARGE, DOCKED MODULES TO A GROUP OF PLATFORMS "FLYING IN FORMATION".
- RANGE IN WEIGHT FROM 12,500 TO 37,000 POUNDS REQUIRING 25 TO 60 FEET STOWED LENGTH.
- EARLY EXPERIMENTAL PLATFORM PLANNED FOR 1987; OPERATIONAL UNITS BY 1992.

SPACE-BASED RADAR

MISSION GOALS

- WOULD PRECLUDE NEED FOR EXPENSIVE UPKEEP OF DEW LINE AND AWACS FLIGHTS
- CAN PROVIDE EARLIER ADVANCE WARNING

BACKGROUND

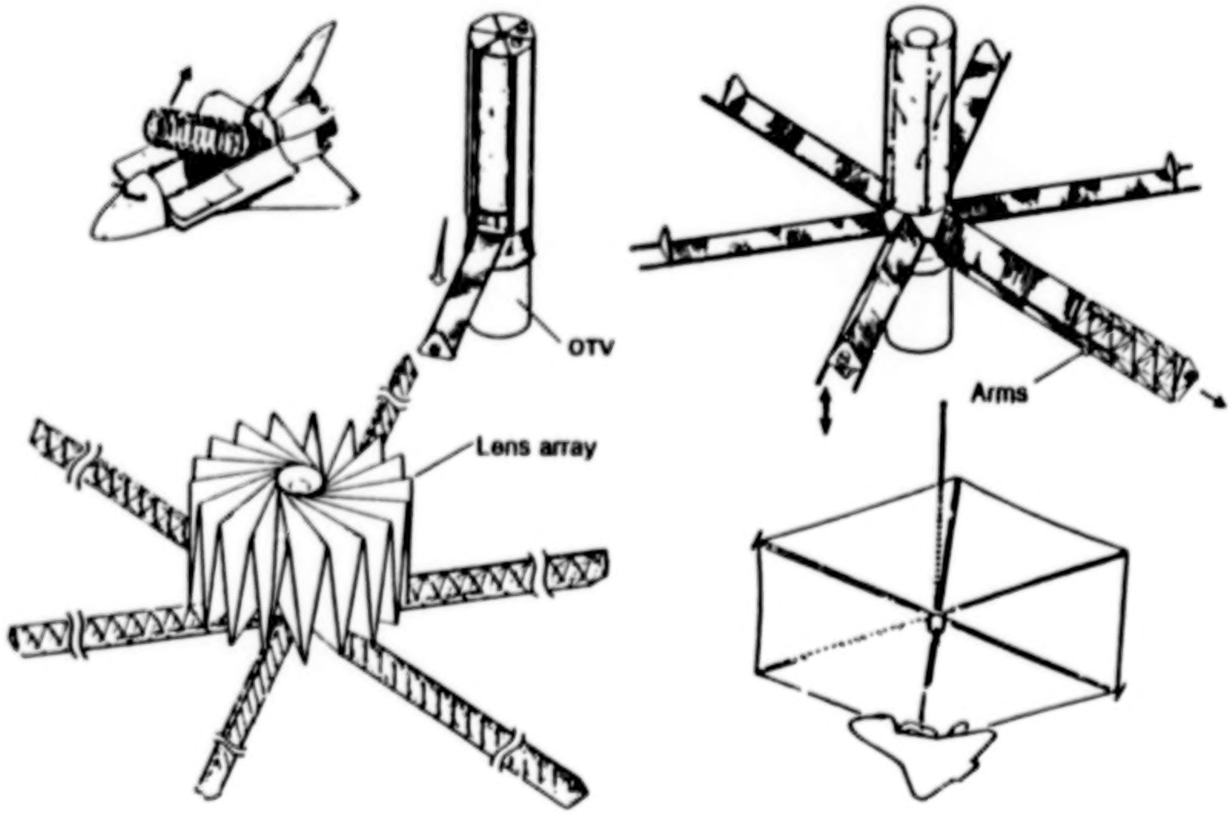
- TEN YEARS OF U.S. NAVY FEASIBILITY STUDIES OF OCEAN SURVEILLANCE SENSORS
- "ON-ORBIT ASSEMBLY" STUDIES FOR SAMSO IN 1978.
- DARPA TECHNOLOGY UNDERWAY, INCLUDING NEW GDC LENS STUDY
- RECENT NASA/MSFC RFP FOR FLIGHT EXPERIMENT OF LARGE DEPLOYABLE ANTENNA

CONCEPTS NOTE: RADAR AND IR SENSORS MAY BE COMBINED IN ONE NETWORK OR ON ONE SPACECRAFT

- POLAR ORBIT
  - ▲ APPROXIMATELY 200 FT DIAMETER GIVES GOOD RESOLUTION
  - ▲ 6 TO 12 SPACECRAFT GIVE COVERAGE
  - ▲ IOC COULD BE AS EARLY AS 1988
  - ▲ EACH SPACECRAFT WEIGHS ~10,000 POUNDS AND REQUIRES ABOUT 25-35 FT STOWED LENGTH
- GEO ORBIT
  - ▲ 300 TO 600 FT DIAMETER NEEDED FOR RESOLUTION
  - ▲ 1 OR 2 SPACECRAFT REQUIRED
  - ▲ IOC PROBABLY WOULD FOLLOW POLAR-ORBIT CONCEPT
  - ▲ EACH SPACECRAFT WEIGHS 15,000-25,000 POUNDS AND REQUIRES ABOUT 60 FT STOWED LENGTH

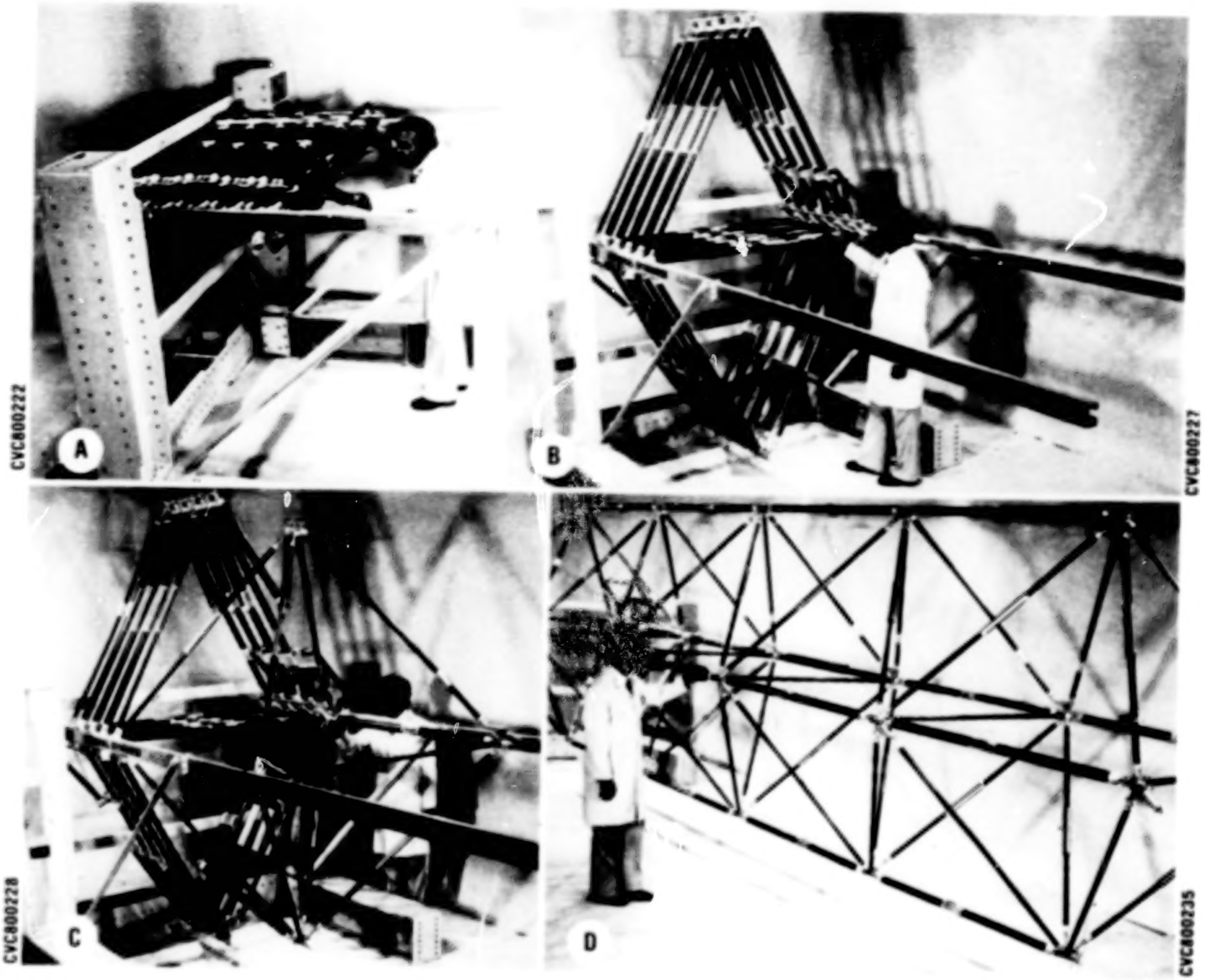
SPACE-BASED RADAR

TETRAHEDRAL TRUSS ARM DEPLOYMENT SEQUENCE



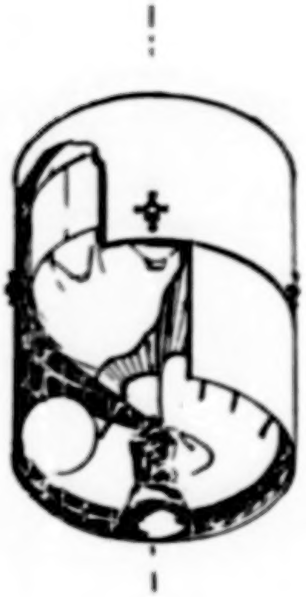
GDC TETRAHEDRAL TRUSS DEMONSTRATION

(GY70/X-30 TUBES)


A
CVC800222
B
CVC800227
C
CVC800228
D
CVC800235

0100-28A

ORBIT TRANSFER VEHICLES/PROPULSION SYSTEMS



RANGE OF REQUIREMENTS IMPOSED ON OTV

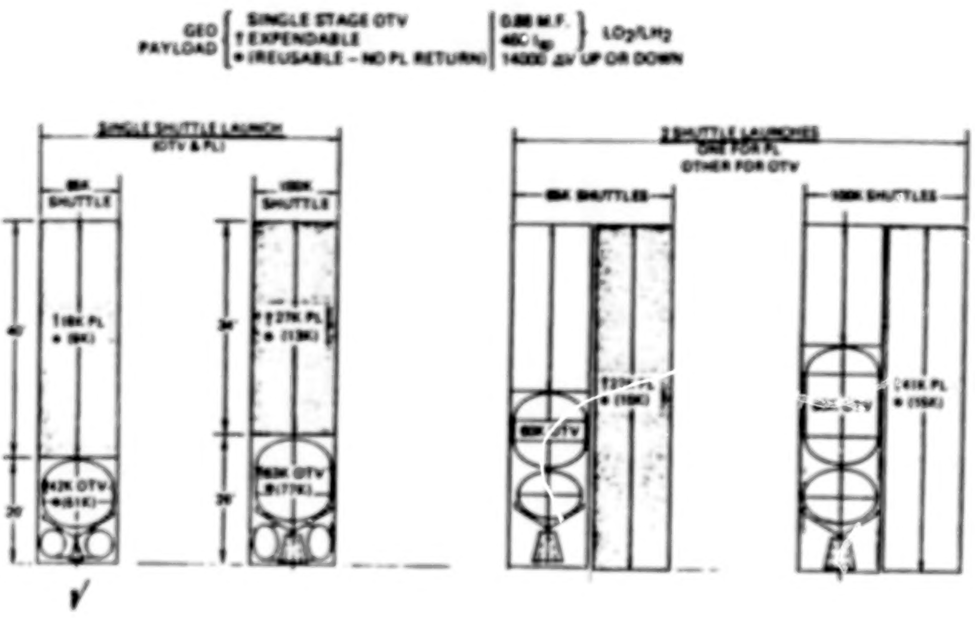
NUMBER	IOC	PAYLOAD WEIGHT	PAYLOAD LENGTH	
13	1987 - 1990	10,000 - 12,500 LB	20-35'	SINGLE SHUTTLE OK
12	1992	15,000 LB	25'	
14	1991 - 1993	15,000 - 25,000 LB	60'	MULTIPLE SHUTTLES REQD
8	1992	25,000 LB	25'	

STARTING IN 1987, THERE ARE [IN THE NASA/MSFC MISSION MODEL FOR OTV STUDY (NAS8-33533)] 25 MISSIONS WHICH BENEFIT FROM LOW THRUST - THAT CAN BE LAUNCHED WITH AN OTV IN A SINGLE SHUTTLE LAUNCH - ENCOURAGING A SHORT OTV.

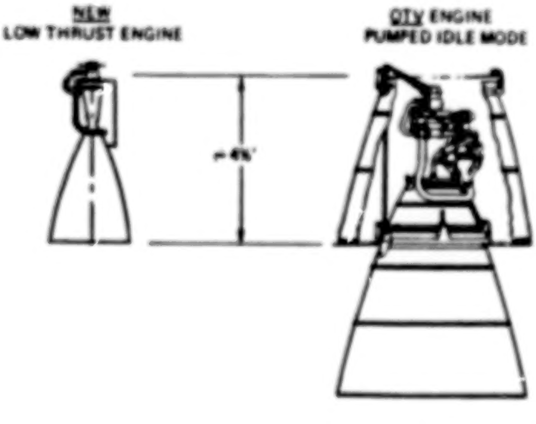
Analysis was conducted for expendable vs. reusable, single stage vs. 2-stage, single vs. multiple Shuttle launches, and 65K vs. 100K Shuttles. The most cost-effective option is the single Shuttle, expendable OTV. This option was selected for primary study.

To obtain the shortest possible stage to allow maximum payload length, the torus LO_2 tank configuration is selected since it is superior to all others (conventional suspended tanks, nested tanks). A savings of 9' in length is realized over conventional tanks.

CANDIDATE OTV CONCEPTS


GEO PAYLOAD
SINGLE STAGE OTV
† EXPENDABLE
• (REUSABLE – NO PL RETURN)
0.88 M.F.
460 Isp
LO2/LH2
14000 ΔV UP OR DOWN
SINGLE SHUTTLE LAUNCH (OTV & PL)
65K SHUTTLE
100K SHUTTLE
†18K PL
• (9K)
40'
20'
42K OTV
34'
†27K PL
• (13K)
2 SHUTTLE LAUNCHES ONE FOR PL OTHER FOR OTV
65K SHUTTLES
100K SHUTTLES
OTV
†41K PL
• (15K)
✓

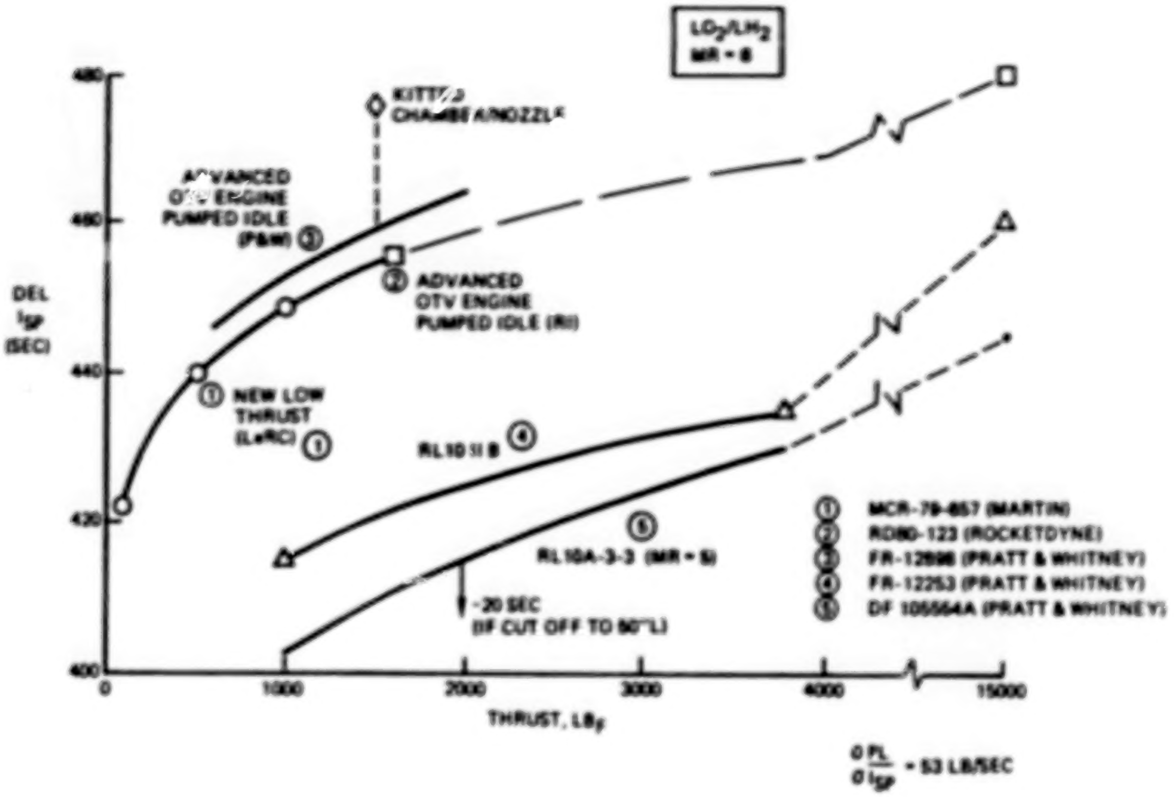
ENGINE OPTIONS


NEW
LOW THRUST ENGINE
OTV ENGINE
PUMPED IDLE MODE
~4½'

		NEW	NEW + KIT*	RL10 II B
THRUST, LB	1500	1500	1500	3500
I_{sp}, SEC	456	456	470	435

*CHAMBER/NOZZLE (SMALLER THROAT, COUNTERFLOW NOZZLE)

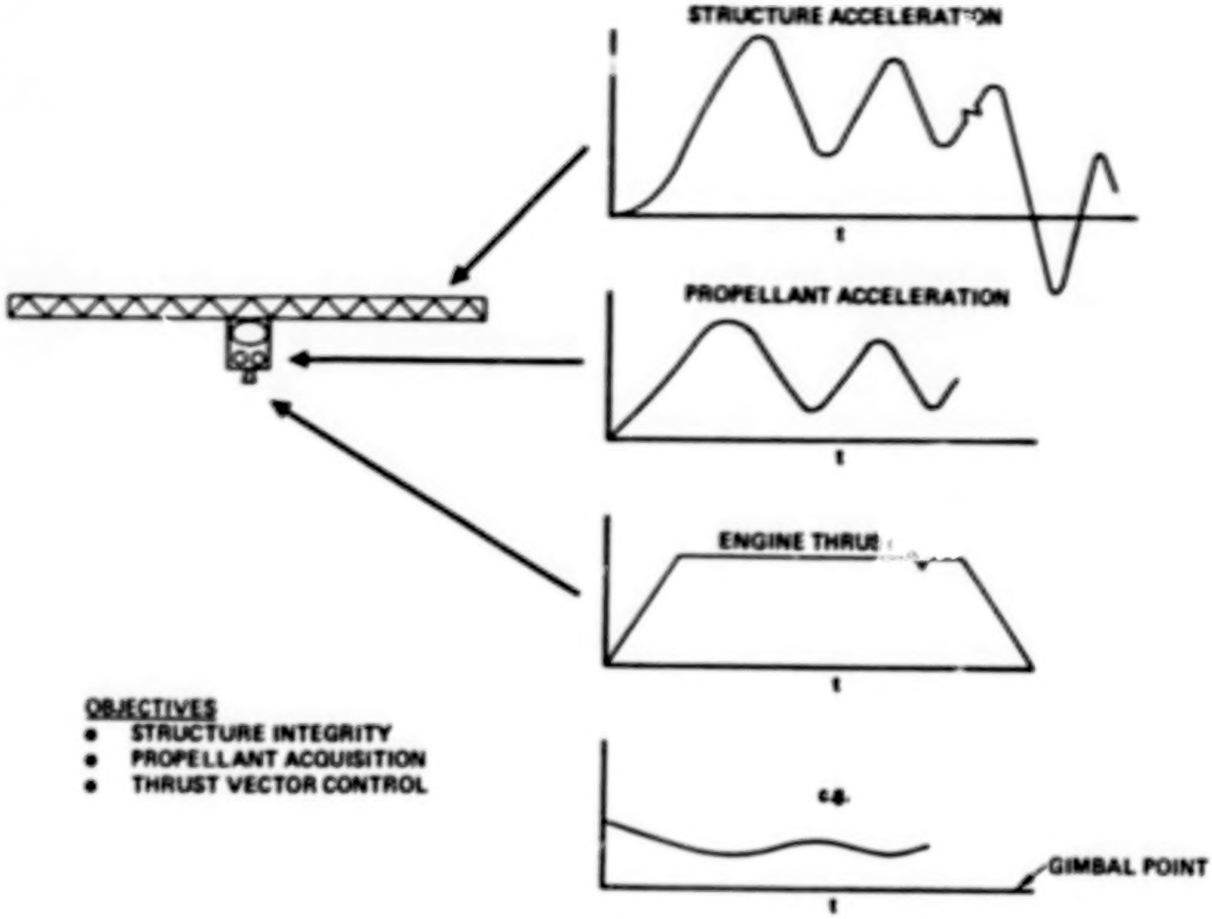
LOW THRUST ENGINE PERFORMANCE



LOW THRUST ENGINE TECHNOLOGY

	NEW LOW THRUST	PUMPED IDLE (OTV ENGINE)
TECHNOLOGY CONCERNS —	— SMALL PUMPS, COOLING, AND PERFORMANCE	— PERFORMANCE AND STABILITY AT 10% THRUST
SIZE —	— SMALLER	— LARGER
WEIGHT —	— LESS	— HEAVIER
REC. COST —	— TBD	— TBD
DEV. COST —	— TBD	— TBD

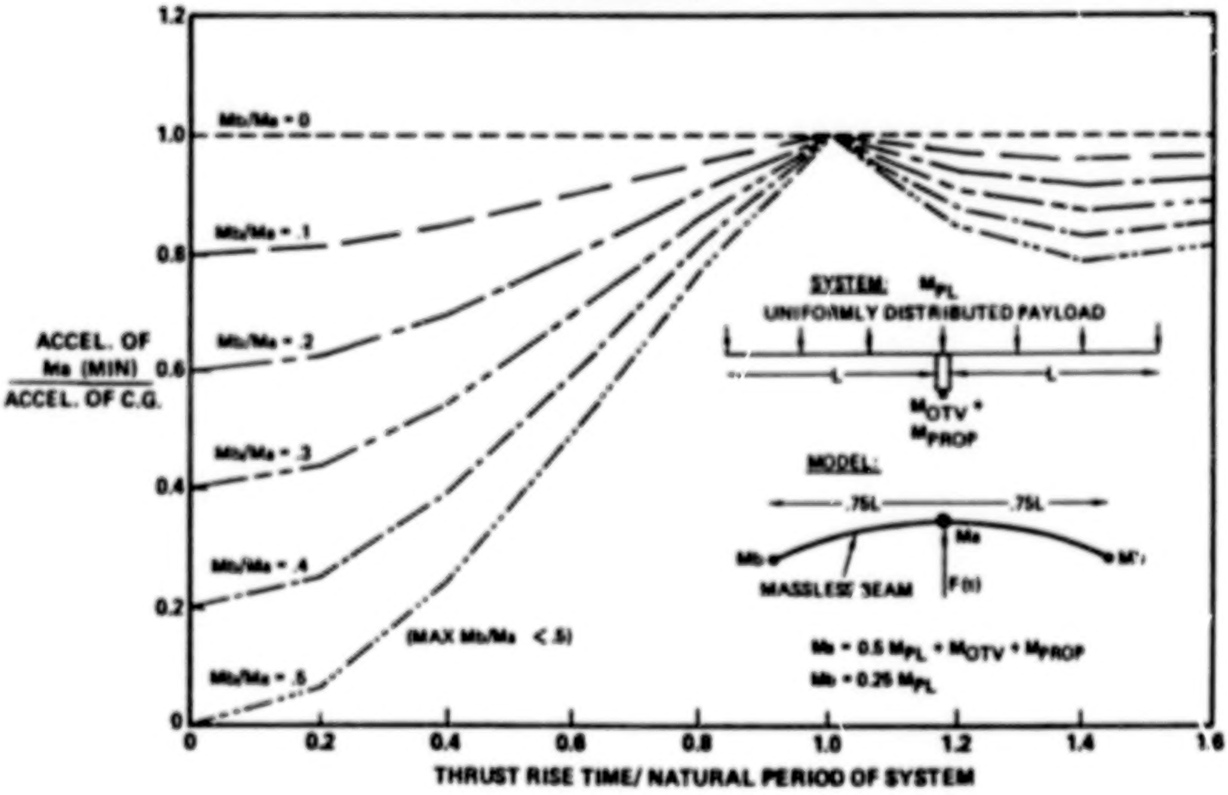
THRUST TRANSIENT INTERACTION


STRUCTURE ACCELERATION
t
PROPELLANT ACCELERATION
t
ENGINE THRUST
t
c.g.
t
GIMBAL POINT

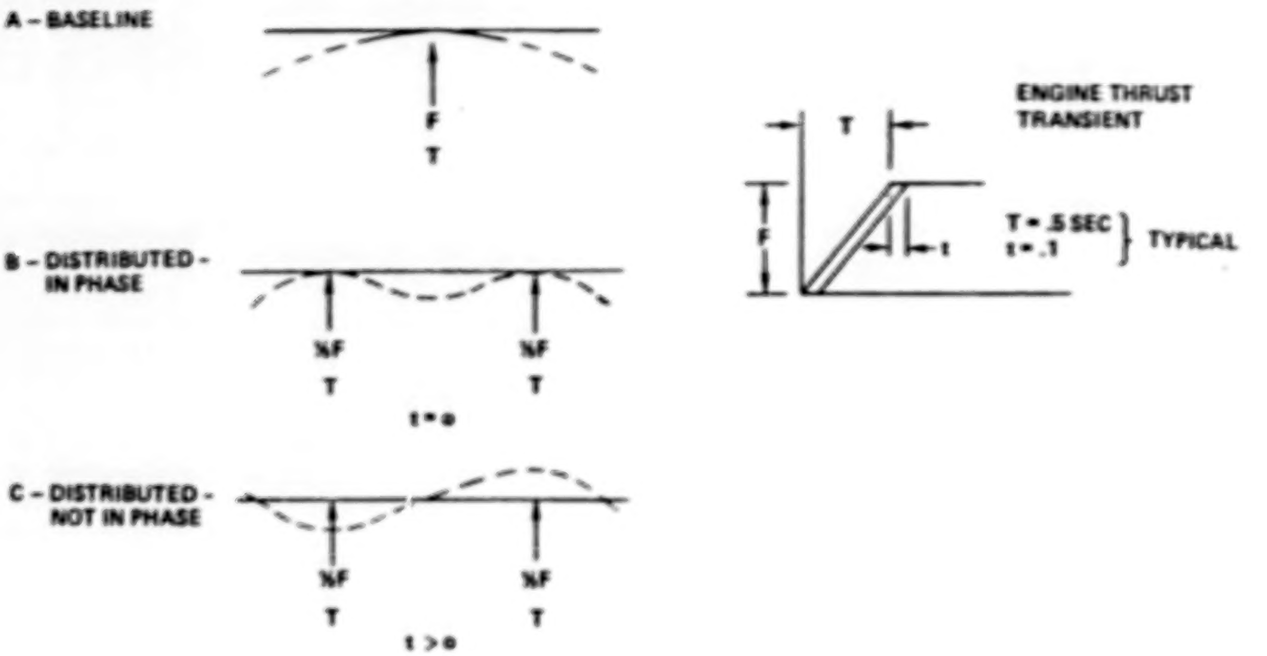
OBJECTIVES
- STRUCTURE INTEGRITY
- PROPELLANT ACQUISITION
- THRUST VECTOR CONTROL

MINIMUM DYNAMIC RESPONSE

(PROPELLANT)



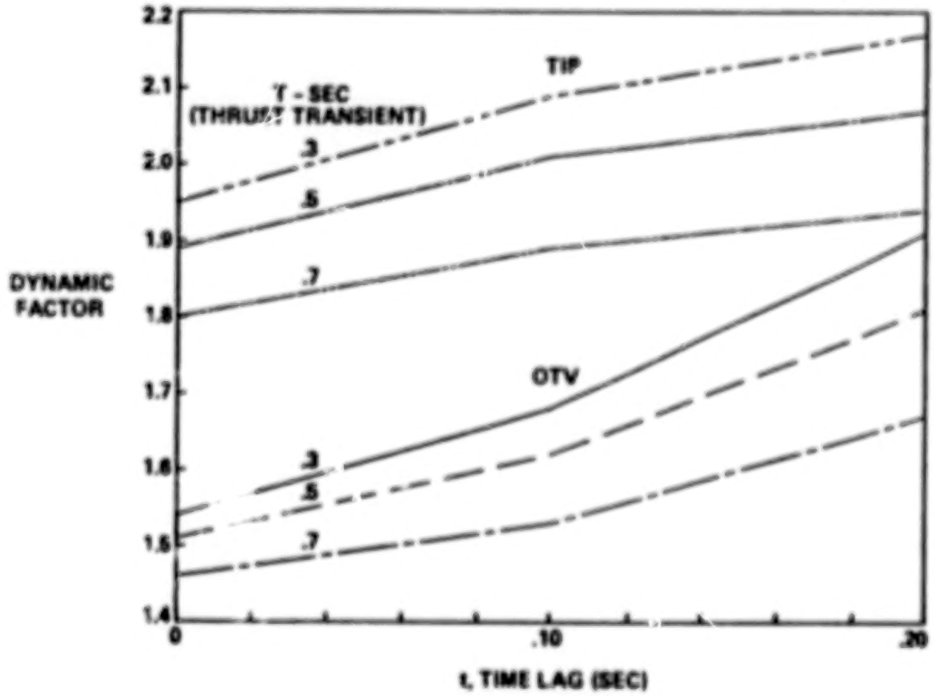
DISTRIBUTED THRUST


A – BASELINE
F
T
B – DISTRIBUTED – IN PHASE
½F
T
½F
T
t = 0
C – DISTRIBUTED – NOT IN PHASE
½F
T
½F
T
t > 0
ENGINE THRUST TRANSIENT
T
F
t
T = .5 SEC
t = .1
TYPICAL

DISTRIBUTED THRUST COMPLICATES OTV/LSS DESIGN/DEPLOYMENT.
DIFFICULTY IN PHASING THRUSTERS CAN INCREASE DYNAMIC LOADING.

DISTRIBUTED THRUST

(EFFECT ON DYNAMIC FACTOR)



The GDC computer program is both a synthesis and optimization program for parametric and trade studies of LSS and OTV configurations operating out of the Shuttle. The program has the following features.

It accepts LSS truss structure material properties, and minimum member size and gage limitations. For purposes of this analysis, graphite composite having an $E = 40 \times 10^6$ psi and an $F_{cy} = 37{,}000$ psi, and aluminum (6061-T6) having an $E = 10^7$ psi and $F_{cy} = 35{,}000$ psi are used. Minimum tube diameter and thickness are 2 and .05 inches, respectively.

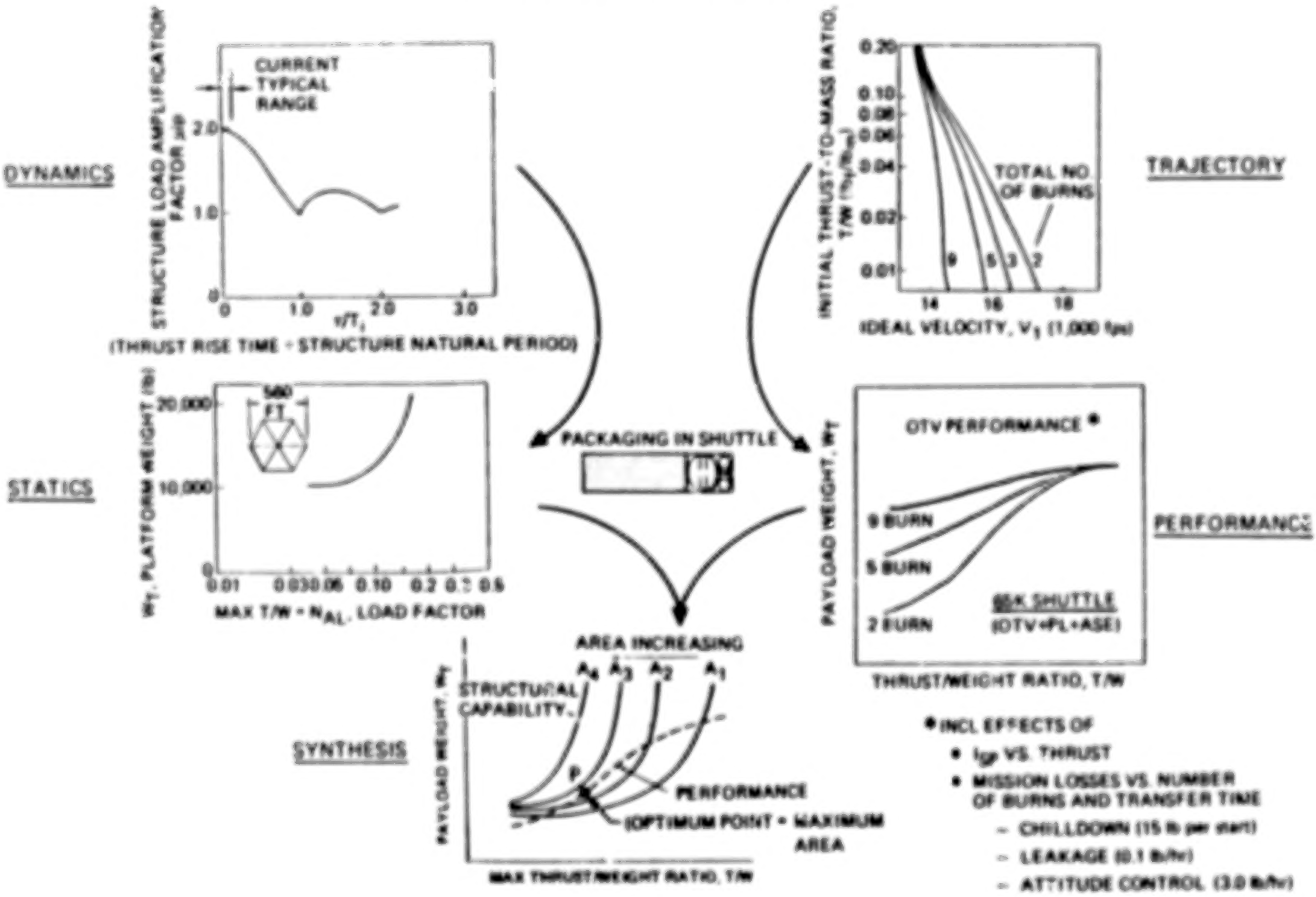
The program accounts for the Shuttle payload weight and volume constraints as well as the configuration of the OTV (i.e., mass fraction and length vs. propellant weight) and its propulsion system I_{sp} vs. thrust characteristics.

The input also includes factors for weight of joints, the LSS hub weight, dynamic amplification factors, and number of burns.

Through an iterative computational process the program computes stowed and deployed sizes as well as structural and mass properties. It checks critical stresses including Euler column buckling of truss member tubes and also radar-array-membrane stresses. If stresses are unacceptable, the tube diameters are first iteratively increased up to the point at which volume limitation constraints are encountered. After this, the tube wall gages are increased as necessary up to the point at which weight limitation constraints are encountered. It then computes OTV length, mass, and performance parameters. To perform these analyses, it must compute ΔV impulse velocity requirements to achieve orbital transfer for the selected input number of burns and initial acceleration.

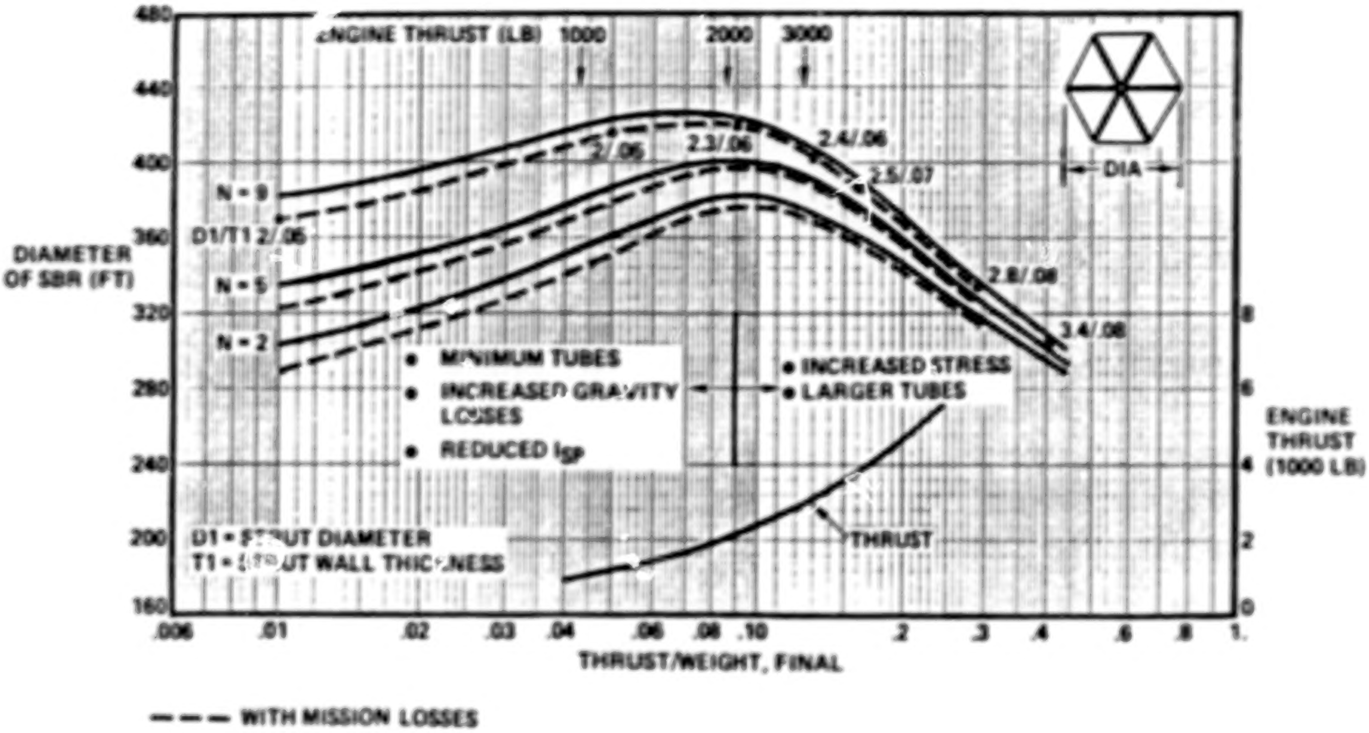
Fit checks are performed to determine for a given T/W and structure size if the payload and volume limitations of the Shuttle are met and if the OTV payload capability matches the actual payload weight. The structure size is then systematically increased until either volume and/or weight limitations are encountered, at which point the maximum LSS size is assumed to have been achieved. The T/W is next increased and the above process is repeated to generate data for LSS size vs T/W. For each T/W all characterizing parameters of the LSS and OTV are computed and printed out along with a factor for the fraction of the total Shuttle cargo bay length utilized. In all cases the full payload capabilities of the Shuttle are used.

PERFORMANCE ANALYSIS

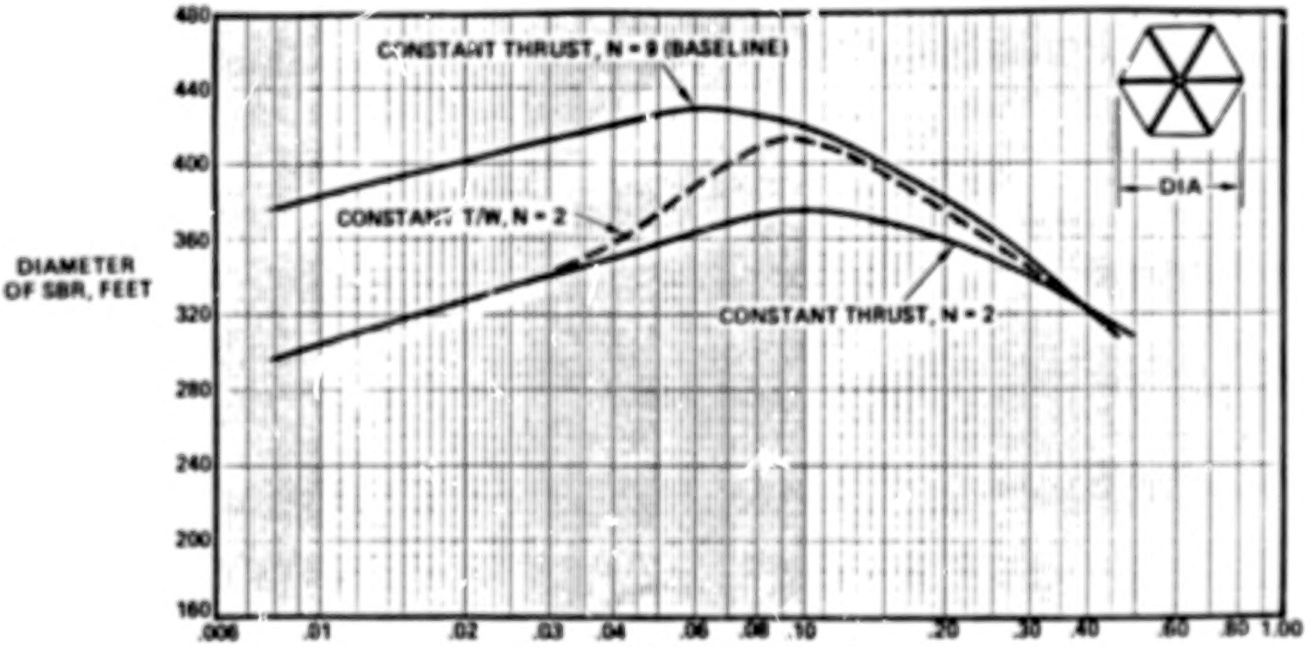

DYNAMICS
STRUCTURE LOAD AMPLIFICATION FACTOR
CURRENT TYPICAL RANGE
2.0
1.0
0
0
1.0
2.0
3.0
τ/T
(THRUST RISE TIME ÷ STRUCTURE NATURAL PERIOD)
TRAJECTORY
INITIAL THRUST-TO-MASS RATIO, T/W
0.20
0.10
0.08
0.06
0.04
0.02
0.01
TOTAL NO. OF BURNS
9
5
3
2
14
16
18
IDEAL VELOCITY, V (1,000 fps)
STATICS
W, PLATFORM WEIGHT (lb)
500 FT
20,000
10,000
0
0.01
0.03
0.05
0.10
0.2
0.3
0.5
MAX T/W = N, LOAD FACTOR
PACKAGING IN SHUTTLE
PERFORMANCE
OTV PERFORMANCE *
PAYLOAD WEIGHT, W
9 BURN
5 BURN
2 BURN
65K SHUTTLE
(OTV+PL+ASE)
THRUST/WEIGHT RATIO, T/W
SYNTHESIS
AREA INCREASING
A4
A3
A2
A1
STRUCTURAL CAPABILITY
PAYLOAD WEIGHT
P
PERFORMANCE
(OPTIMUM POINT = MAXIMUM AREA
MAX THRUST/WEIGHT RATIO, T/W

* INCL EFFECTS OF
- I_{sp} VS. THRUST
- MISSION LOSSES VS. NUMBER OF BURNS AND TRANSFER TIME
  - CHILLDOWN (15 lb per start)
  - LEAKAGE (0.1 lb/hr)
  - ATTITUDE CONTROL (3.0 lb/hr)
  - POWER (0.5 lb/hr)
  - BOILOFF (MAX) (1.0 lb/hr)

EFFECT OF ENGINE THRUST & NUMBER OF BURNS ON SIZE OF SBR-A



EFFECT OF CONSTANT ACCELERATION (VARIABLE THRUST) ON SIZE OF SBR-A



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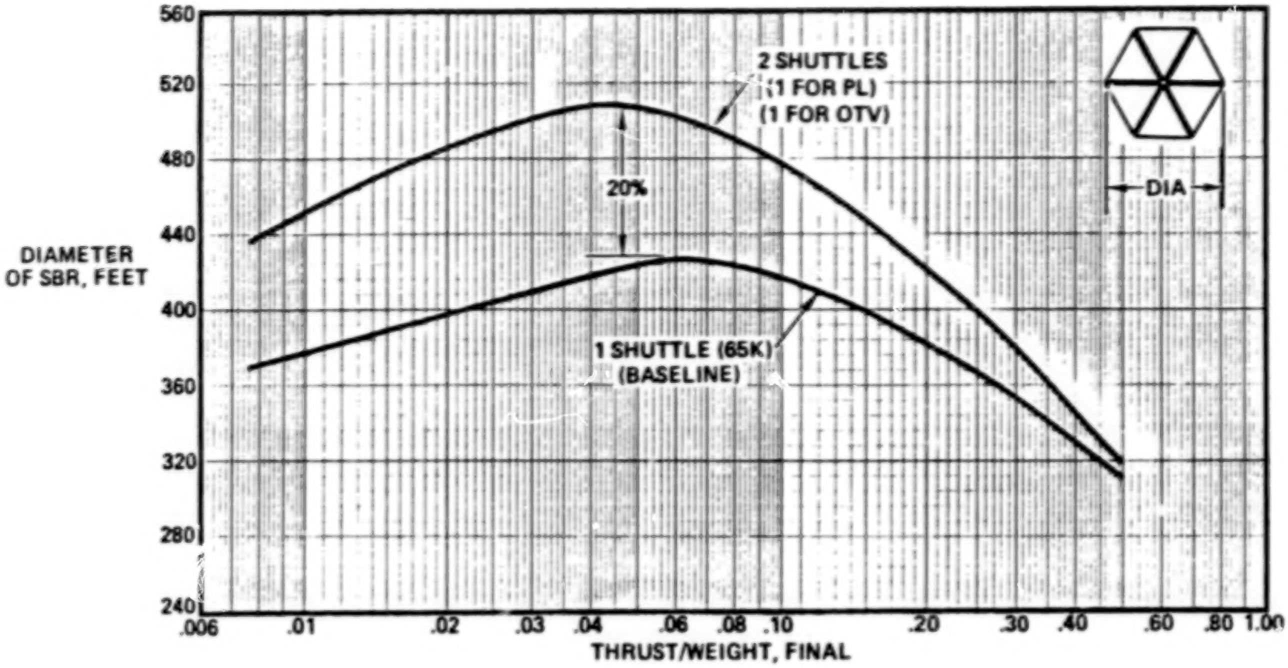
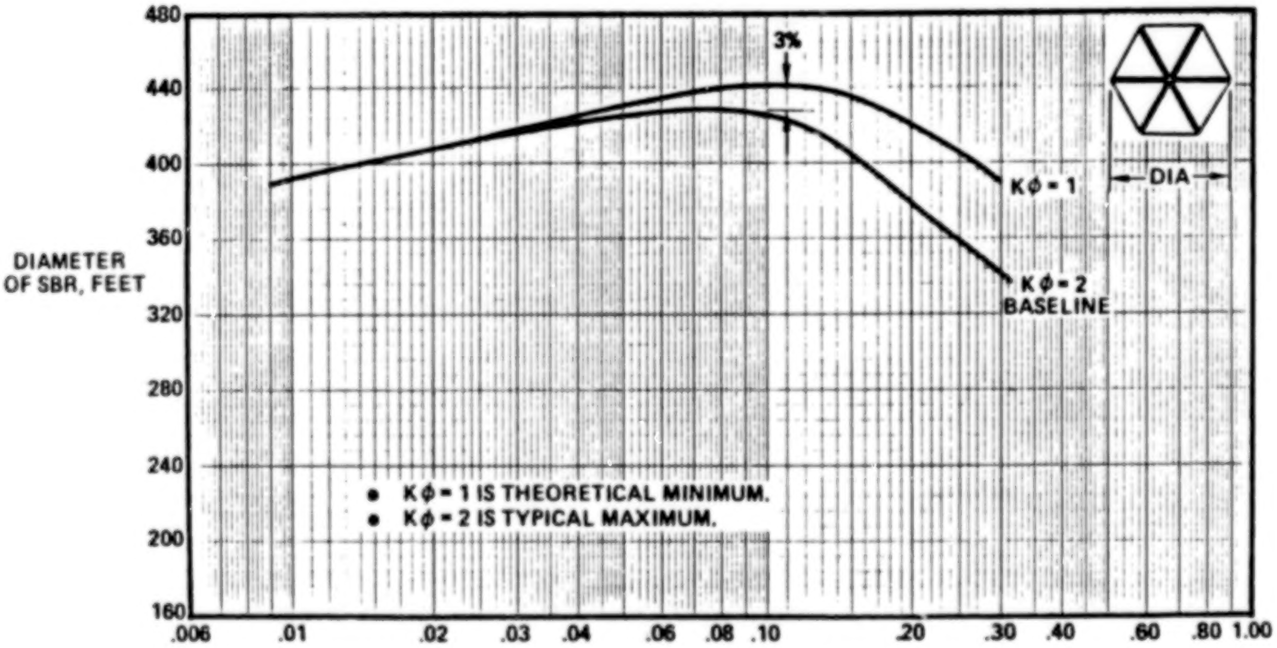
EFFECT OF NUMBER OF SHUTTLES ON SIZE OF SBR-A

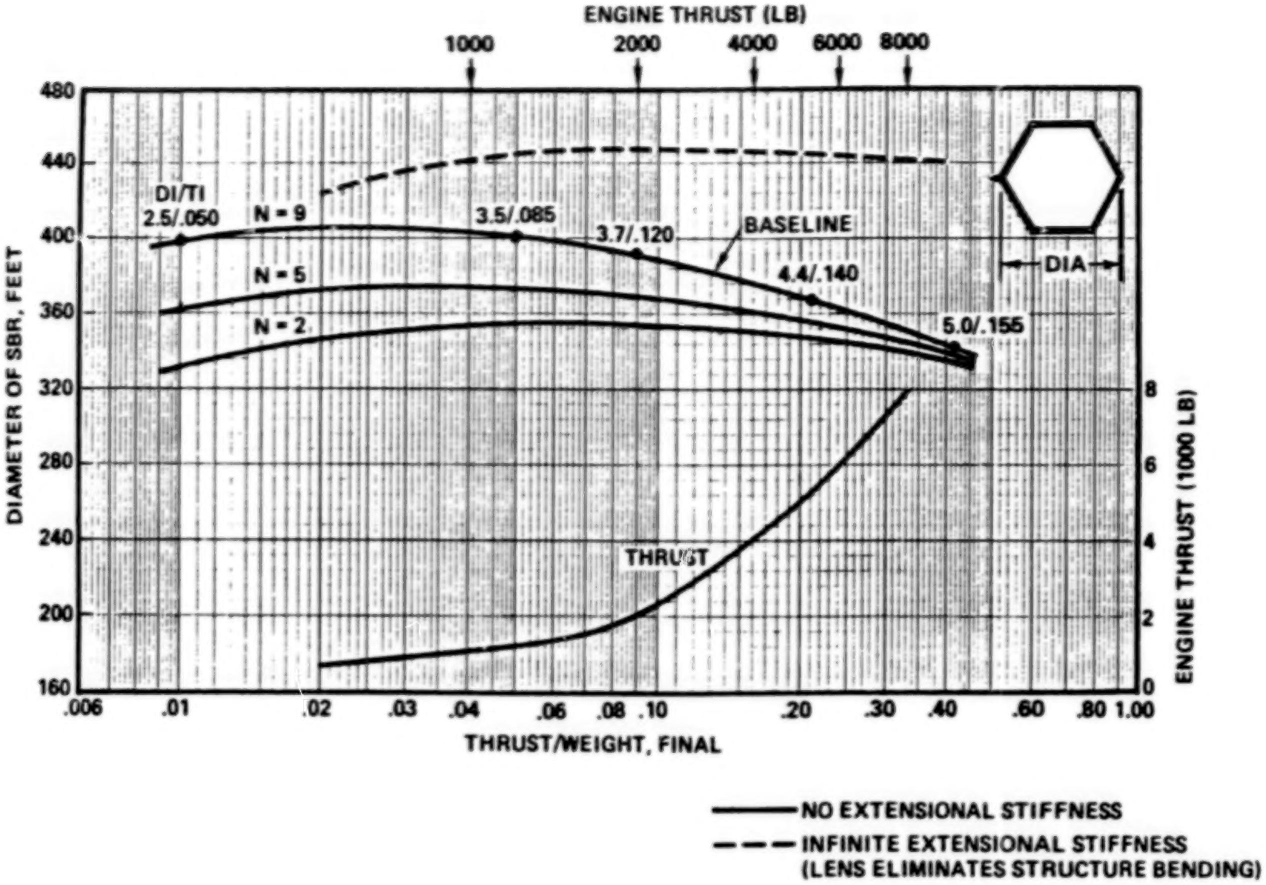


EFFECT OF DYNAMIC FACTOR (Kϕ) ON SIZE OF SBR-A

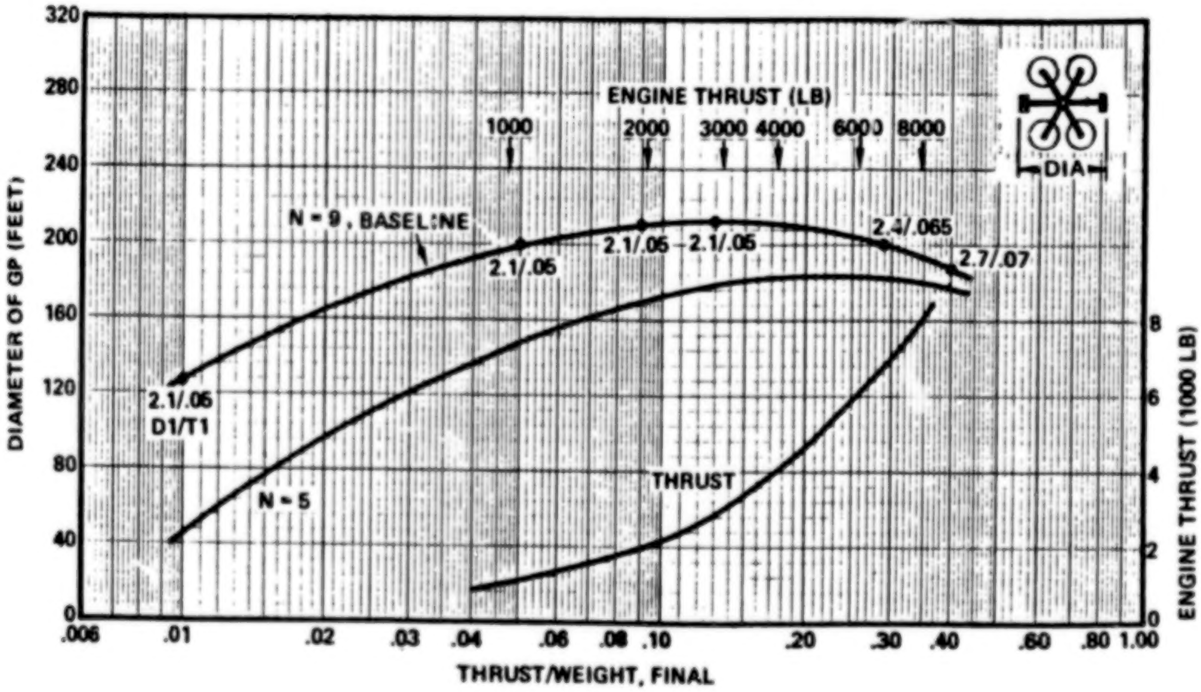


- Kϕ = 1 IS THEORETICAL MINIMUM.
- Kϕ = 2 IS TYPICAL MAXIMUM.

EFFECT OF ENGINE THRUST & NUMBER OF BURNS ON SIZE OF SBR-R



EFFECT OF ENGINE THRUST & NUMBER OF BURNS ON SIZE OF GEOPLATFORM



INTERACTION RESULTS

SUMMARY

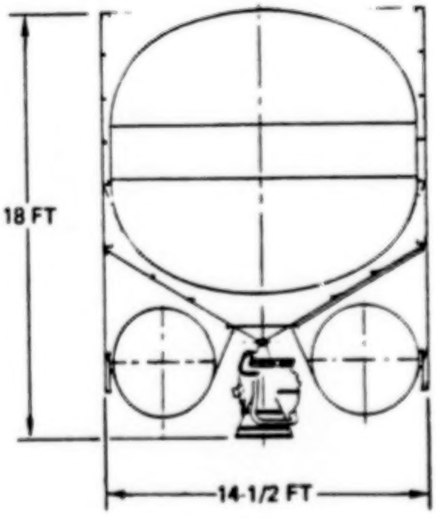
- OPTIMUM THRUST

SBR-A	2000 LB_F	(MOST SENSITIVE)
SBR-R	1000 LB_F	(LEAST SENSITIVE)
GP	3000 LB_F	

- THRUST TRANSIENT NOT A CONCERN
- CONSTANT THRUST (9-BURN) BEST

1500 LB_F THRUST LEVEL SELECTED FOR BASELINE

BASELINE DESIGN DEFINITION

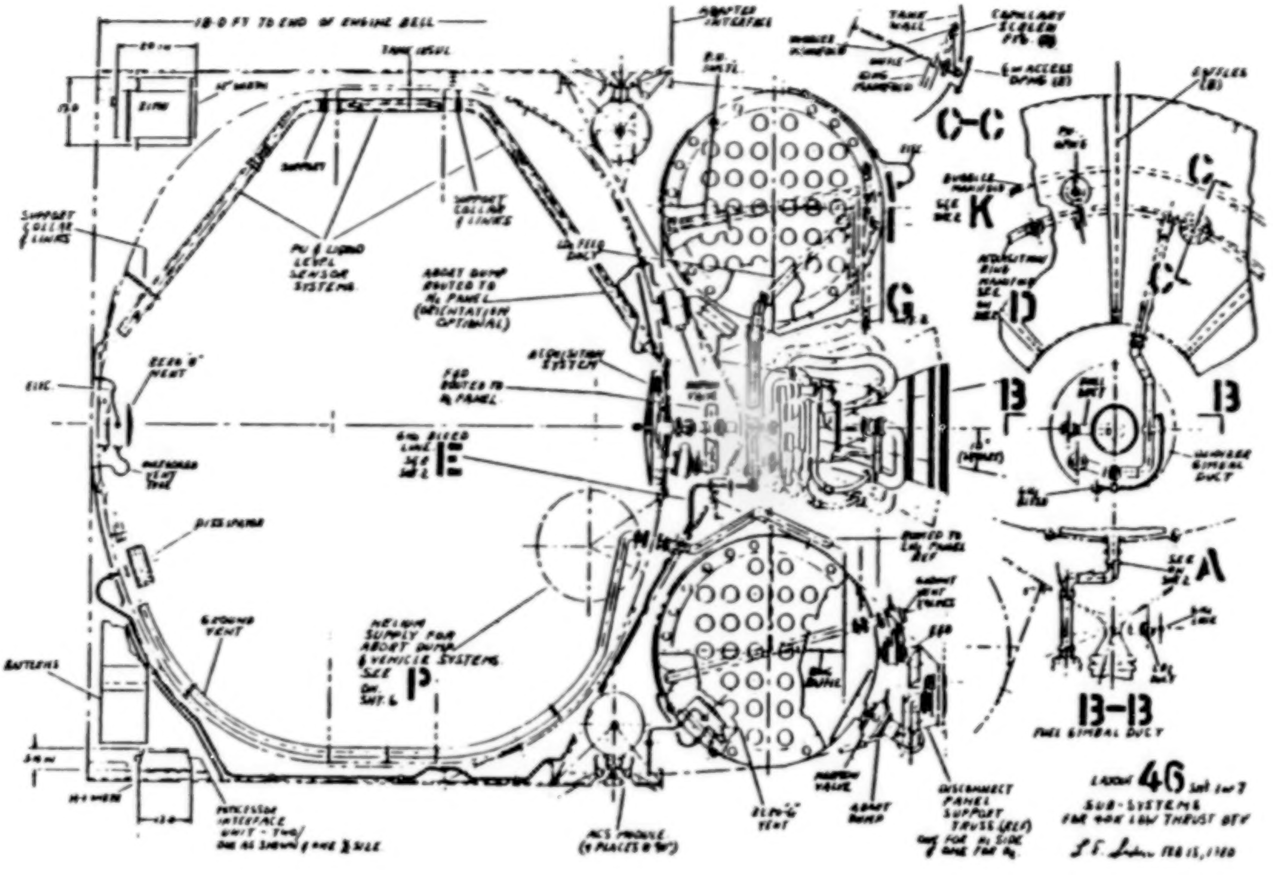


(DESIGNED FOR 3 g IN SHUTTLE).

EXPENDABLE LOW THRUST OTV
(38K PROPELLANT @ MR = 6)

- PUMP FED (1.5K) ENGINE
  - ▲ ENGINE-MOUNTED/DRIVEN PUMPS (NO VEHICLE – MOUNTED BOOST PUMPS)
  - ▲ 16 PSIA MIN INLET PRESSURE
  - ▲ NPSH
    - LO_2 – 1 PSI
    - LH_2 – 0.5 PSI
  - ▲ AUTOGENOUS H_2 BLEED
- COMPOSITE STRUCTURE
- ALUMINUM TANKS
- PROPELLANT ACQUISITION
  - ▲ PARTIAL SETTLING
  - ▲ SCREENS
- MLI TANK INSULATION (15 LAYERS)
- PRESSURIZATION
  - ▲ HELIUM PRE-PRESS; O_2 RUN
  - ▲ AUTOGENOUS H_2 RUN
- ZERO-G VENT/MIXER
- FILL AND DRAIN }
- 300 SEC ABORT DUMP } THROUGH SIDES OF ORBITER
- N_2H_4 ATTITUDE CONTROL
- FUEL CELL POWER (1 KW)
- MISSION
  - ▲ 40–HR ORBITER C/O
  - ▲ 24–HR TRANSFER
  - ▲ 9 BURNS
  - ▲ 5 HR BURN TIME

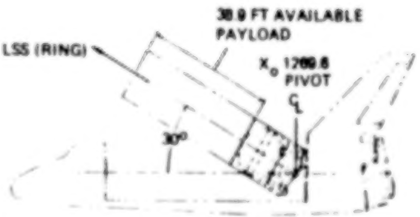
BASELINE
LOW THRUST OTV


18-0 FT TO END OF ENGINE BELL
C-C
C
K
D
G
B
E
P
A
B-B
FUEL GIMBAL DUCT
LAYOUT 46
SUB-SYSTEMS FOR 40K LOW THRUST OTV

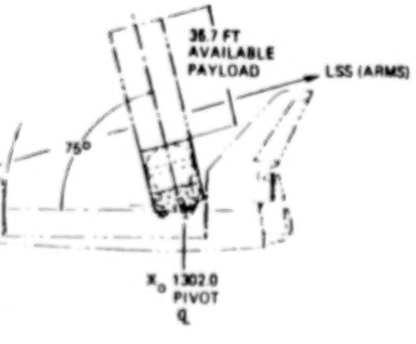
NOTE: SHORT RL10 USED TO DEFINE INTERFACES

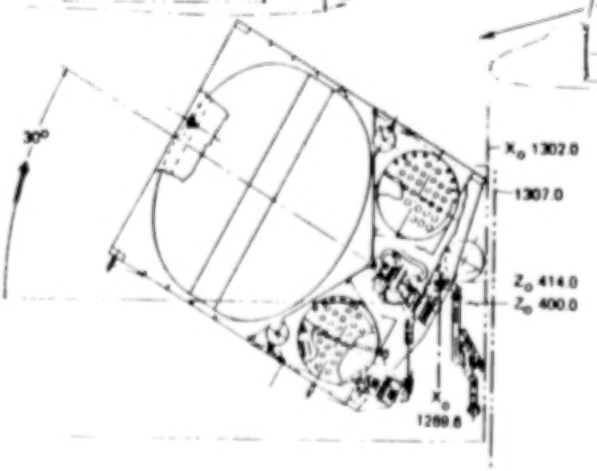
DEPLOYMENT

A -

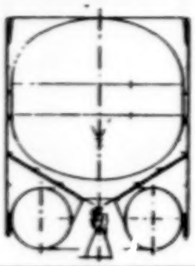

LSS (RING)
38.9 FT AVAILABLE PAYLOAD
X_o 1289.8 PIVOT
℄
30°

B -


35.7 FT AVAILABLE PAYLOAD
LSS (ARMS)
75°
X_o 1302.0 PIVOT
℄

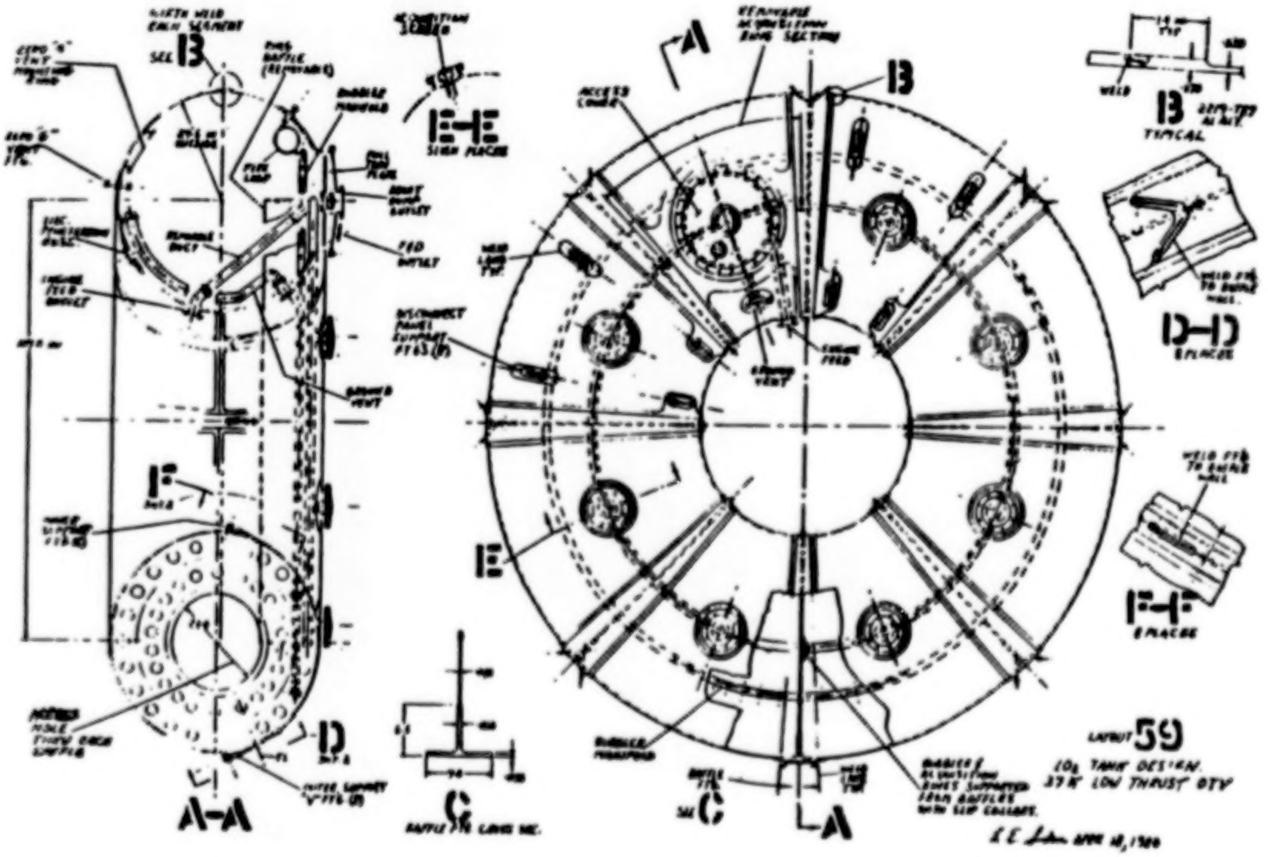

30°
X_o 1302.0
1307.0
Z_o 414.0
Z_o 400.0
X_o 1289.8

WEIGHT SUMMARY
LOW THRUST OTV

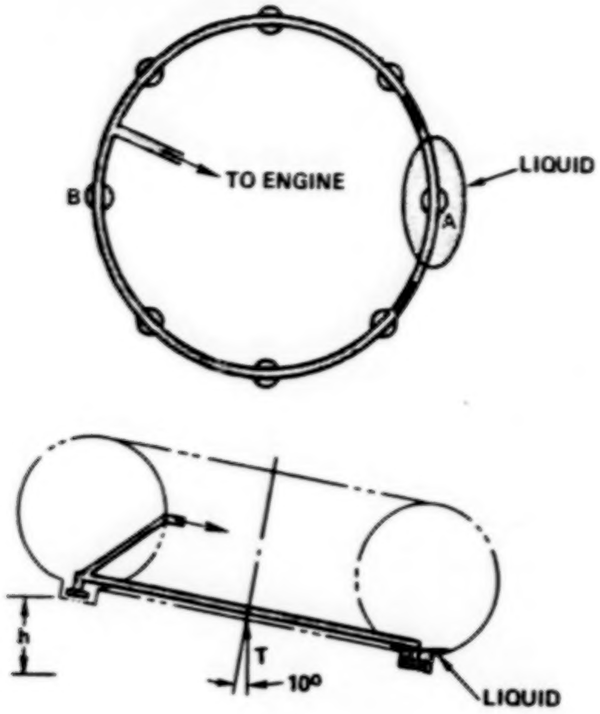


WEIGHT DATA (LB)	
STRUCTURE	2,177
THERMAL CONTROL	536
MAIN PROPULSION	762
ATTITUDE CONTROL	206
AVIONICS	396
ELECTRICAL POWER	380
CONTINGENCY	668
TOTAL DRY WEIGHT	5,124
RESIDUALS	382
RESERVES	430
BURNOUT WEIGHT	5,936
INFLIGHT LOSSES	319
MAIN IMPULSE PROPELLANT	37,434
ACS PROPELLANT (INCL DISPOSAL ΔV)	551
STAGE TOTAL WEIGHT	44,240
PAYLOAD TO GEOSYNCHRONOUS ORBIT (MAX)	15,760
STAGE PLUS PAYLOAD WEIGHT	60,000
AIRBORNE SUPPORT EQUIPMENT	5,000
TOTAL LAUNCH WEIGHT	65,000
MASS FRACTION	0.856

TORUS LO_2 TANK DESIGN

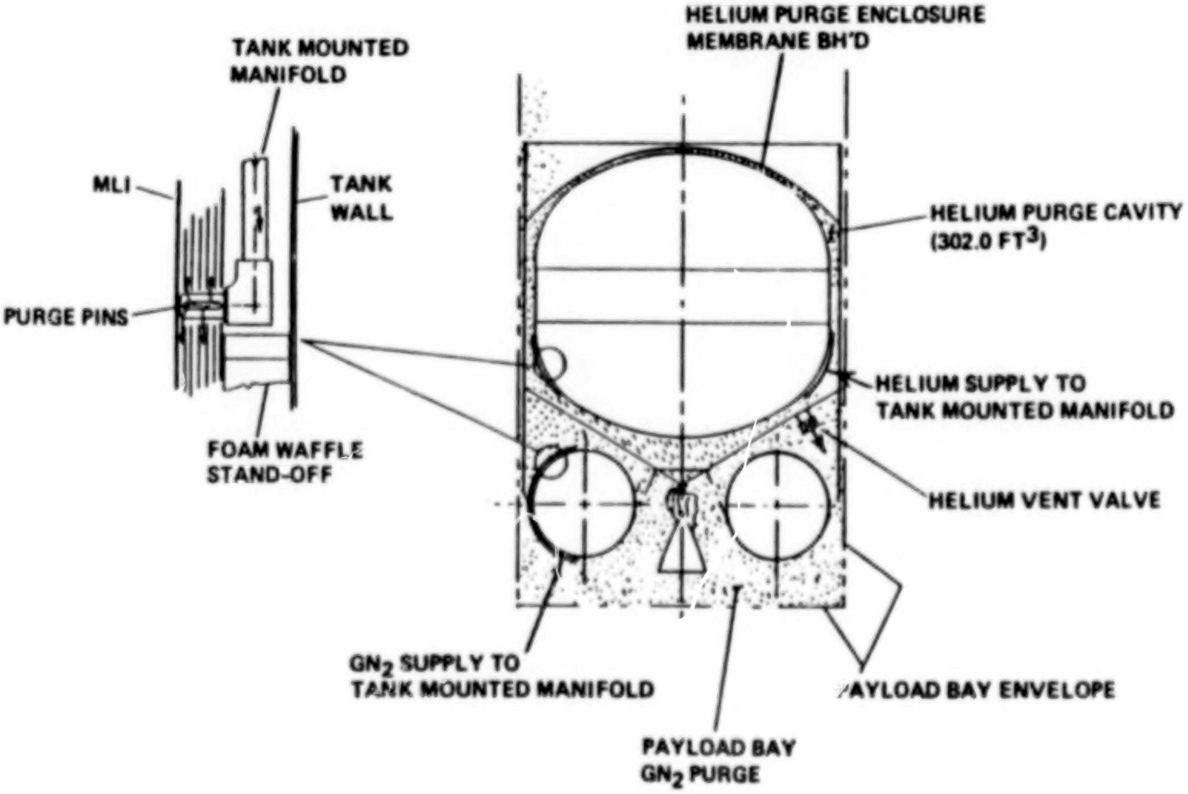

SEC B
E-E
A
B
ACCESS COVER
B TYPICAL
D-D
F
E
F-F
8 PLACES
D
A-A
C
SEC C
A
LAYOUT 59
LO2 TANK DESIGN
37K LOW THRUST OTV

LO_2 ACQUISITION WITH THRUST MISALIGNMENT

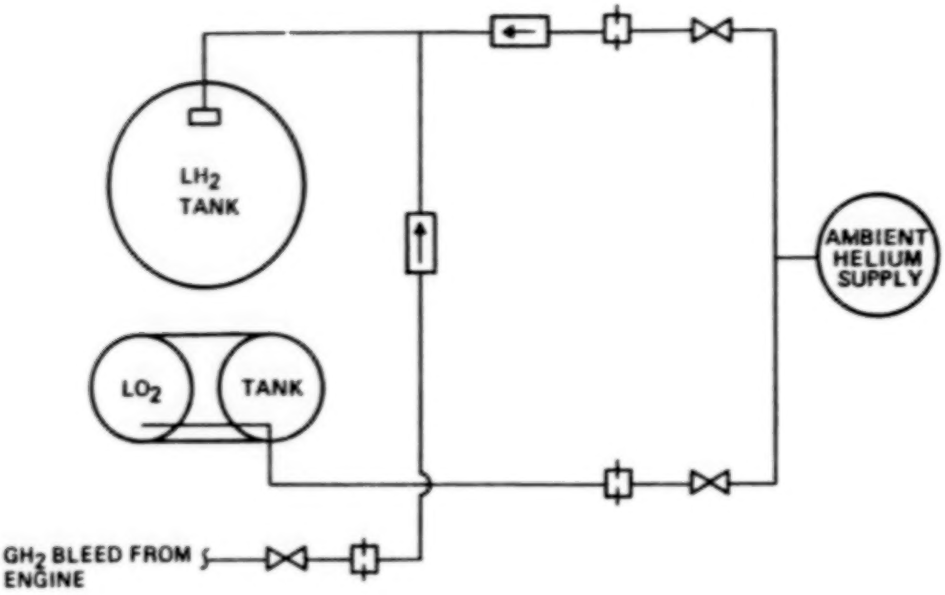

TO ENGINE
LIQUID
B
A
h
T
10°
LIQUID

TORUS PROPELLANT ACQUISITION DEVICE MINIMIZES RESIDUALS WITH C.G. MISALIGNMENT.

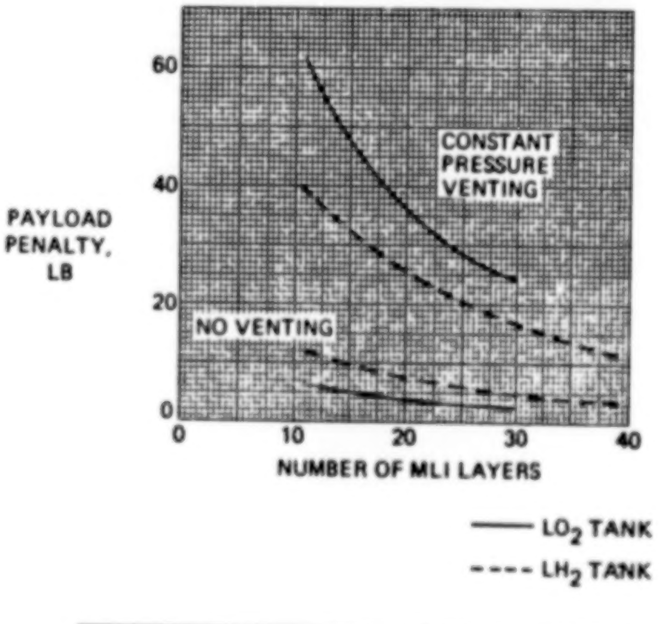
PURGE SYSTEM ENCLOSURE


HELIUM PURGE ENCLOSURE MEMBRANE BH'D
TANK MOUNTED MANIFOLD
MLI
TANK WALL
HELIUM PURGE CAVITY ($302.0 FT^3$)
PURGE PINS
HELIUM SUPPLY TO TANK MOUNTED MANIFOLD
FOAM WAFFLE STAND-OFF
HELIUM VENT VALVE
GN_2 SUPPLY TO TANK MOUNTED MANIFOLD
PAYLOAD BAY ENVELOPE
PAYLOAD BAY GN_2 PURGE

PROPELLANT TANK PRESSURIZATION SYSTEM


LH$_2$ TANK
LO$_2$ TANK
AMBIENT HELIUM SUPPLY
GH$_2$ BLEED FROM ENGINE

40-HR CHECKOUT (PAYLOAD PENALTY)



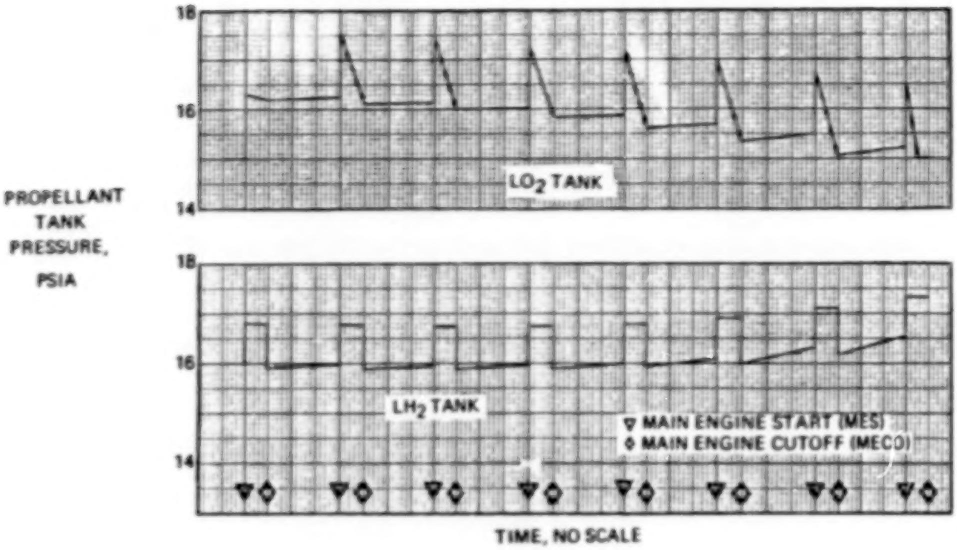
NO VENT CONDITION – PRESSURES DO NOT EXCEED DESIGN VALUES (SET BY ABORT DUMP REQUIREMENTS)

19 PSIA – LH$_2$
25 PSIA – LO$_2$

ZERO "G" VENT/MIXER

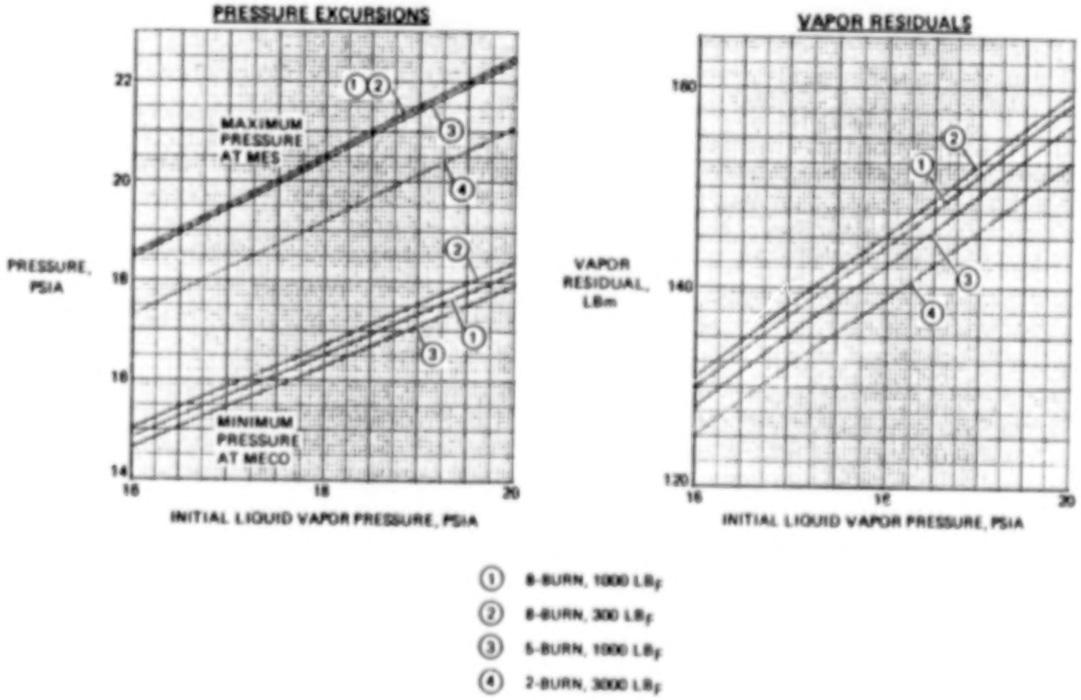
PAYLOAD PENALTY FOR 40-HR CHECKOUT AT LEO IS MINIMIZED WITH NO VENT OPTION

PROPELLANT TANK PRESSURE HISTORIES FOR EIGHT-BURN OTV MISSION



- LO_2 TANK PRESSURIZED WITH HELIUM FOR ENGINE START AND ENGINE BURN
- LH_2 TANK PRESSURIZED WITH HELIUM FOR ENGINE START; AUTOGENOUS PRESSURIZATION FOR ENGINE BURN
- ENGINE NPSP REQUIREMENT
  - 1.0 PSI LO_2
  - 0.5 PSI LH_2

OTV MISSION PARAMETERS INFLUENCE UPON LO_2 TANK



LO_2 TANK VAPOR RESIDUALS OR PRESSURES LITTLE AFFECTED BY MISSION – ENGINE THRUST OR NUMBER OF BURNS

TECHNOLOGY DEVELOPMENT

TECHNOLOGY DEVELOPMENT SHOULD BE UNDERTAKEN FOR ENGINE AND VEHICLE SYSTEMS
ESTIMATED INVESTMENT NEEDED

TORUS TANK	—	\$3-5M	FABRICATION AND TEST
PROPELLANT ACQUISITION	—	\$1M	
INSULATION	—	\$0.5M	
LOW THRUST ENGINE	—	\$3-7M	BOTH NEW LOW THRUST AND PUMPED IDLE
TOTAL		\$7-14M	

> TECHNOLOGY INVESTMENT IS NEEDED FOR LOW THRUST OTV

CONCLUSIONS

THIS STUDY HAS DEFINED AN OPTIMIZED LOW THRUST OTV CONFIGURED SPECIFICALLY FOR ORBIT TRANSFER OF LARGE SPACE SYSTEMS - WITH THE FOLLOWING CONCLUSIONS:

ENGINE FOR OPTIMUM LOW THRUST VEHICLE

- VERY LOW THRUST (< 1K) NOT REQUIRED.
- 1 - 5K THRUST RANGE APPEARS OPTIMUM.
- THRUST TRANSIENT NOT A CONCERN.
- THROTTLING NOT WORTHWHILE.
- MULTIPLE THRUSTERS COMPLICATE OTV/LSS DESIGN AND AGGRAVATE LSS LOADS.
- NEW LOW THRUST ENGINE HAS ADVANTAGES OVER OTV PUMPED IDLE ENGINE.

OPTIMUM VEHICLE FOR LOW ACCELERATION MISSIONS

- SINGLE SHUTTLE LAUNCH (LSS AND EXPENDABLE OTV) MOST COST-EFFECTIVE AND LEAST RISK (ADEQUATE FOR 25 LSS MISSIONS).
- MULTIPLE SHUTTLES INCREASE LSS DIAMETER 20%.
- SHORT OTV NEEDED WHICH REQUIRES USE OF TORUS TANK
- PROPELLANT TANK PRESSURES/VAPOR RESIDUALS LITTLE AFFECTED BY THRUST LEVEL OR NUMBER OF BURNS.

RECOMMENDATIONS

FURTHER STUDY

- REVISE RESULTS AS NEW MISSION AND SPACECRAFT DATA BECOME AVAILABLE (ESPECIALLY AS THE GEOPLATFORM DESIGN EVOLVES).
- REEVALUATE STUDY RESULTS AS LeRC LOW THRUST ENGINE STUDIES PRODUCE DESIGN CONCEPTS AND COST DATA.
- COORDINATE WITH OTV STUDY (NAS8-33533 FOLLOW-ON).
- FURTHER EVALUATE BENEFITS OF DEPLOYING LSS AT LEO VS GEO.
- EVALUATE HOW CENTAUR (WITH IDLE MODE) COULD SATISFY REQUIREMENTS.
- ESTIMATE THE POINT AT WHICH ADVANCED ELECTRIC OTV (FAST TRANSFER/MPD) MIGHT REPLACE LOW THRUST CHEMICAL PROPULSION.

TECHNOLOGY

- UNDERTAKE TECHNOLOGY DEVELOPMENT FOR THE ENGINES AND VEHICLE SUBSYSTEMS (LOW THRUST OPTIONS. TORUS TANK, ACQUISITION, INSULATION).

LOW-THRUST VEHICLE CONCEPT STUDIES

George R. Smolak
NASA Lewis Research Center

LOW THRUST VEHICLE CONCEPT STUDIES

- OBJECTIVES
- SCHEDULE
- PACKAGING STUDIES
    - SHUTTLE CARGO BAY CONSTRAINTS
    - LOW THRUST ENGINES
        - PROFILES
        - PERFORMANCE
    - LARGE SPACE FRAME
        - CONCEPT
        - WEIGHT
    - LOW THRUST VEHICLES STOWED IN SHUTTLE
    - LSS PAYLOAD CAPABILITY
    - WEIGHT DISTRIBUTION
- CONCLUSIONS

OBJECTIVES

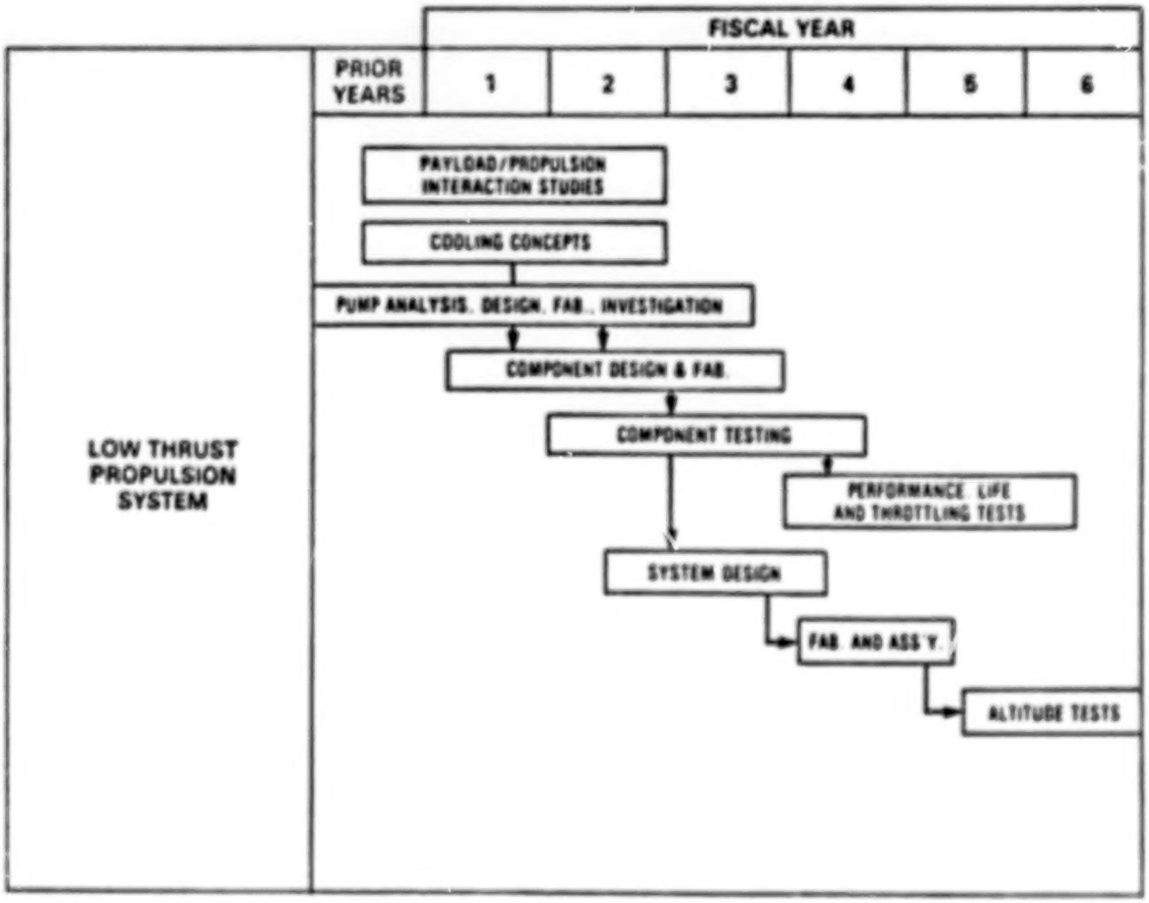
- PROVIDE ANALYTICAL TOOLS TO DEFINE PROPULSION SYSTEM PERFORMANCE, WEIGHT, SIZE, ETC.

- DEVELOP PACKAGING CONCEPTS FOR LSS MISSION PROPULSION AND PAYLOAD SYSTEMS

ORBITAL TRANSFER VEHICLE PROPULSION SCHEDULE

The NASA Lewis low-thrust, vehicle concept studies are part of the NASA-OAST orbit transfer vehicle propulsion program. These studies are a portion of the effort identified as payload/propulsion interaction studies in the schedule chart. Dr. Priem addressed the overall schedule in his introductory remarks on the Low Thrust Propulsion Technology Program.

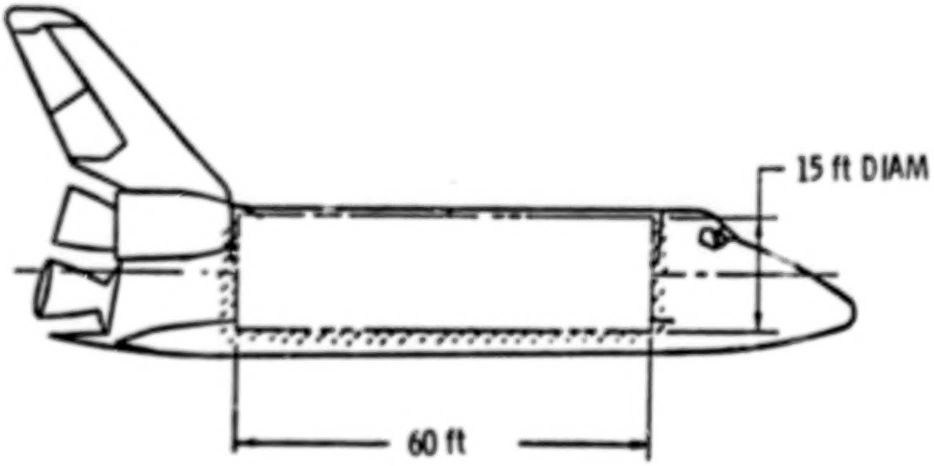
ORBITAL TRANSFER VEHICLE PROPULSION SCHEDULE


FISCAL YEAR
PRIOR YEARS
1
2
3
4
5
6
LOW THRUST PROPULSION SYSTEM
PAYLOAD/PROPULSION INTERACTION STUDIES
COOLING CONCEPTS
PUMP ANALYSIS, DESIGN, FAB., INVESTIGATION
COMPONENT DESIGN & FAB.
COMPONENT TESTING
PERFORMANCE, LIFE AND THROTTLING TESTS
SYSTEM DESIGN
FAB. AND ASS'Y.
ALTITUDE TESTS

SHUTTLE CARGO BAY CONSTRAINTS

A number of Shuttle cargo bay constraints are important in the design of payload systems. The stowed vehicle (payload) must fit within the bay volume (15 ft. diameter by 60 ft. length) and must not exceed 65,000 pounds gross weight. Other major constraints arising from a ride in the Shuttle bay are; vibration, shock, acoustic and thermal environments and center-of-gravity location.

SHUTTLE CARGO BAY CONSTRAINTS

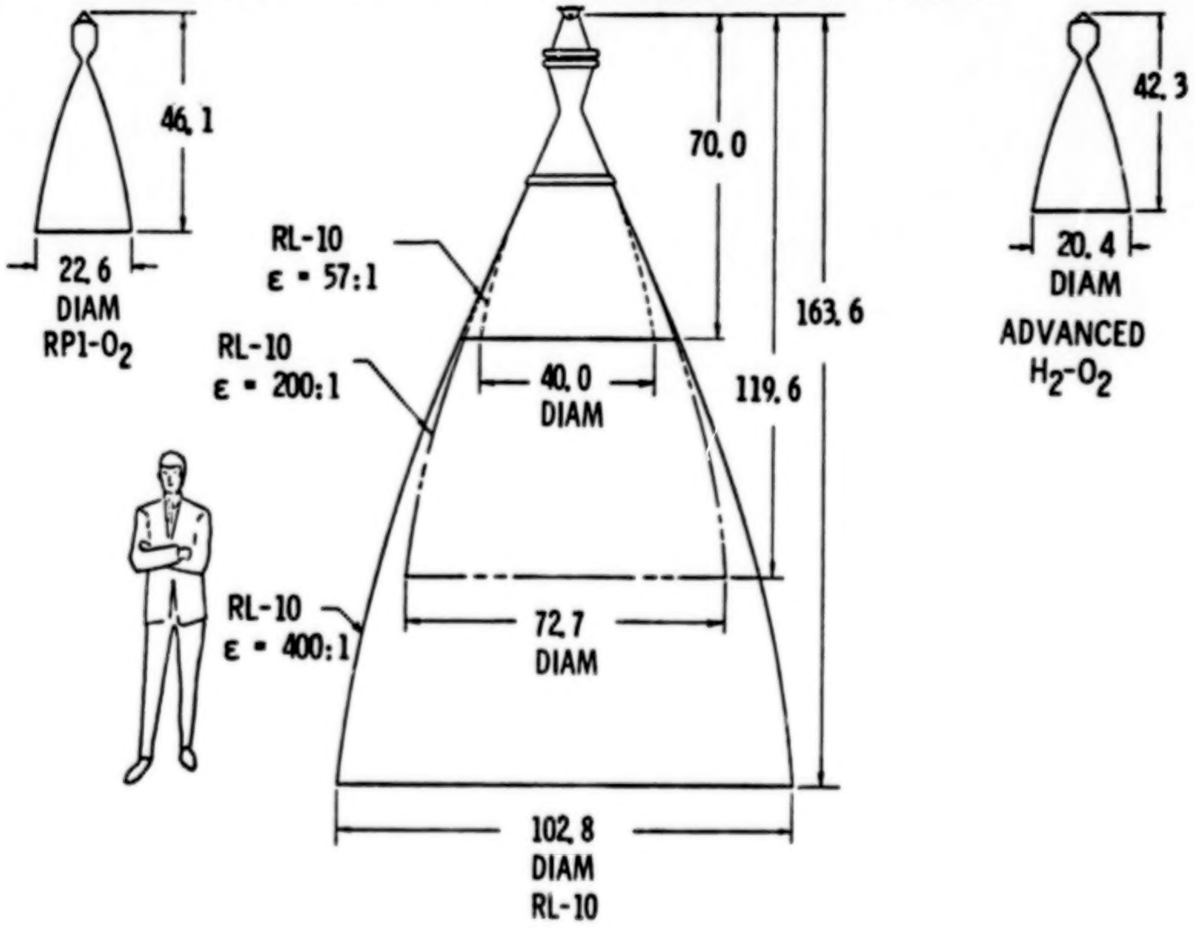


CARGO BAY LAUNCH WEIGHT CAPABILITY • 65 000 lb

APPROXIMATE SIZES OF LOW THRUST CHEMICAL ROCKET ENGINES

Additional constraints on the design of Shuttle payloads are imposed by the physical dimensions of typical low-thrust chemical rocket engines. The engine profiles include; (a) the Pratt and Whitney RL-10 (center sketch) with three different expansion ratio nozzles (57:1, 200:1, and 400:1). All dimensions on the chart are inches. The man shown is drawn to the same scale as the rocket engines. A large savings in engine length can be made if a significant length of the nozzle can be designed to retract. In the upper right portion of the chart is shown an advanced H_2-O_2 engine profile. A low thrust RP1-O_2 engine profile is shown in the upper left.

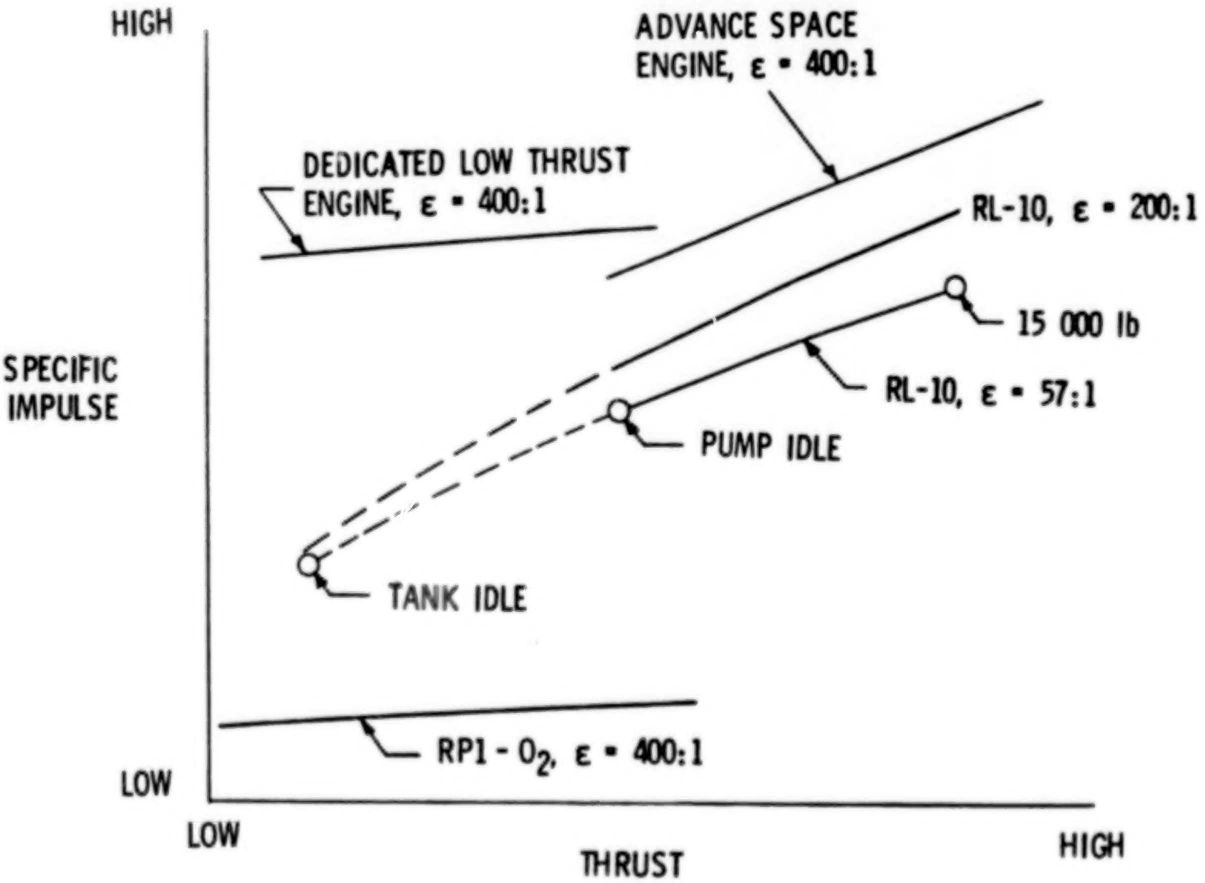
APPROXIMATE SIZES OF LOW THRUST CHEMICAL ROCKET ENGINES


46.1
22.6
DIAM
RP1-O_2
RL-10
ε = 57:1
RL-10
ε = 200:1
RL-10
ε = 400:1
70.0
40.0
DIAM
119.6
163.6
72.7
DIAM
102.8
DIAM
RL-10
42.3
20.4
DIAM
ADVANCED
H_2-O_2

RELATIVE PERFORMANCE OF VARIOUS ENGINES

This chart shows the relative performance of several candidate low thrust chemical rocket engines. Relative specific impulse is shown as a function of thrust for several engines (RL-10 family, Advanced Space Engine, dedicated low thrust H_2-O_2 engine, and RP1-O_2 engine). In its Centaur version, the Pratt and Whitney RL-10 engine produces 15,000 pounds of thrust. The same engine in idle modes produces much lower thrust (1500 pounds during pump idle mode and about 200 pounds during tank idle mode). However, the specific impulse is lower during idle mode operation. The Advanced Space Engine has a favorable high specific impulse, but its thrust is too high for "low thrust" missions. A dedicated low-thrust H_2-O_2 engine is needed. It should have a specific impulse almost as high as the Advanced Space Engine. The dedicated engine would thus offer a significant performance advantage compared to the RL-10 and RP1-O_2 engines.

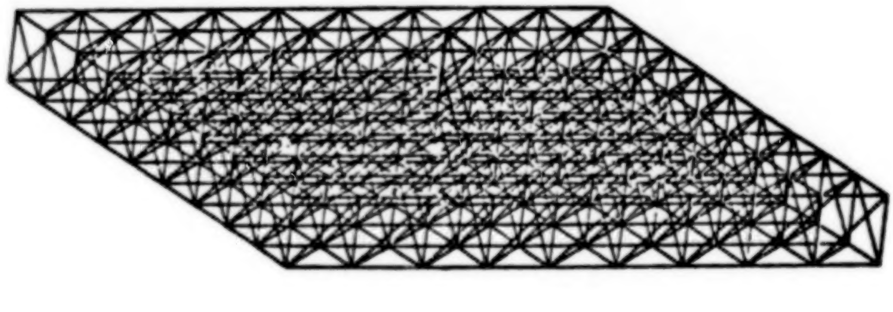
RELATIVE PERFORMANCE OF VARIOUS ENGINES



LARGE SPACE FRAME PLATFORM CONCEPT

Many large space structures have been proposed in the literature. The large deployed space frame shown in the chart is typical of one family of these large structures. Dimensions of these structures generally run hundreds of feet in length and width and up to about 50 feet in depth. Since they are deployed from the Shuttle bay, the structures must be stowable. Materials generally proposed for these structures are epoxy-graphite thin wall tubes, joined by end fittings and wires.

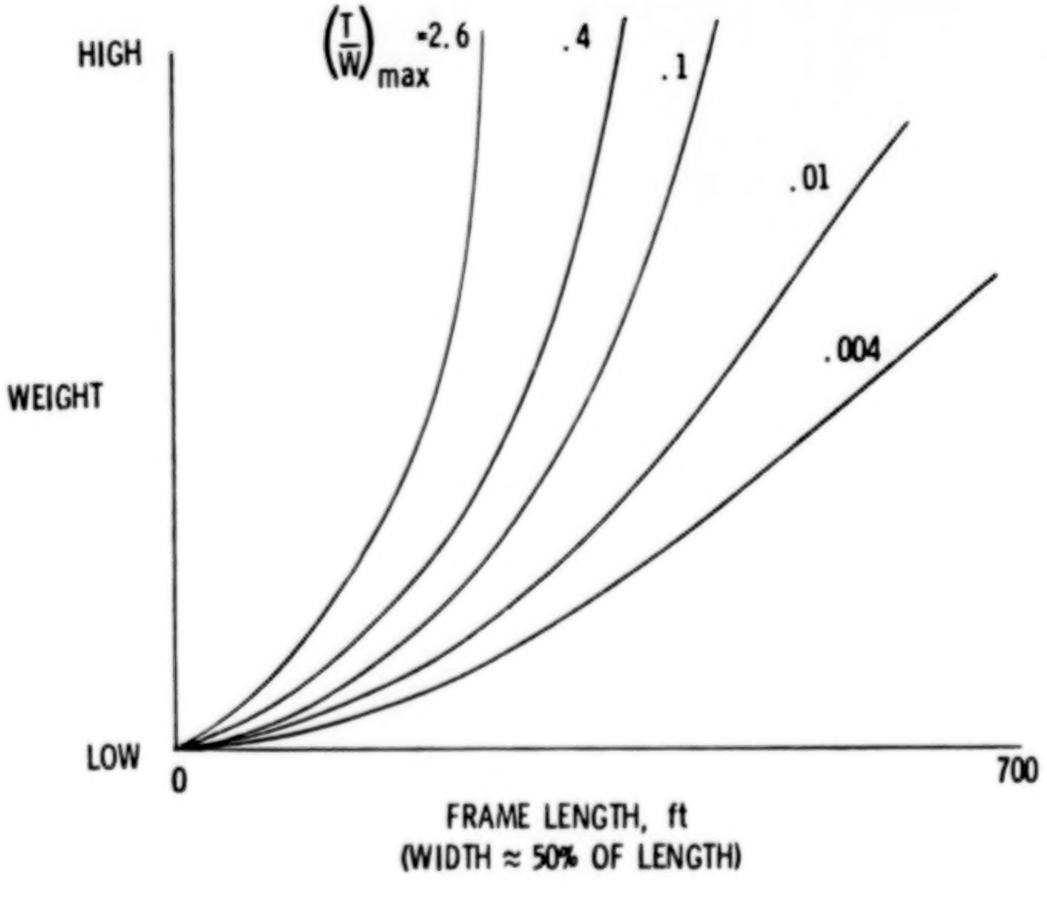
LARGE SPACE FRAME PLATFORM CONCEPT



WEIGHT OF LARGE SPACE FRAME PLATFORMS

This chart shows the relative weight of deployed large space frames of the type shown in the previous chart. Frame weight is shown as a function of frame length for a variety of thrust-to-weight ratios. Frame width has been assumed equal to about 50% of frame length. For the desired frame lengths of many hundreds of feet, the chart indicates that the frame weight will be low for low thrust-to-weight ratios, but very high for high thrust-to-weight ratios. The weights shown are minimum for on-orbit control stiffness. Clearly, low thrust-to-weight ratios are desirable to maximize space frame deployed dimensions.

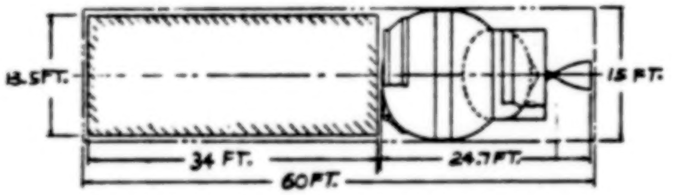
WEIGHT OF LARGE SPACE FRAME PLATFORMS



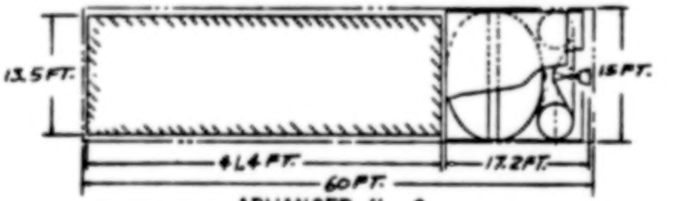
LSS SYSTEMS STOWED IN SHUTTLE BAY

The next chart shows a number of large space structure (LSS) systems (including propulsion systems) as they would appear when stowed in the Shuttle cargo bay. The four top configurations shown (large space frames with; modified Centaur using the RL-10 engine in the tank idle mode, advanced H_2-O_2 engine, RP1-O_2 engine and advanced H_2-O_2 engine with same space frame as RP1-O_2 engine) represent the results of recent NASA-Lewis in-house packaging studies. The goal of the studies was to design compact, light-weight propulsion modules having high specific impulse so that the volume available for the stowed space frame was maximized. Each of the top three LSS stowed systems has a 65,000 pound gross weight. The bottom configuration has the same LSS stowed system as the RP1-O_2 example but weighs less than 65,000 pounds. All of the stowed frames have a density close to 2.5 pounds per cubic foot. The system using the advanced H_2-O_2 engine has the largest space frame capability and the RP1-O_2 engine system has the least payload carrying capability. Each propulsion system was sized to raise its respective deployed payload from low earth orbit (LEO) to geosynchronous orbit in several days with several burns.

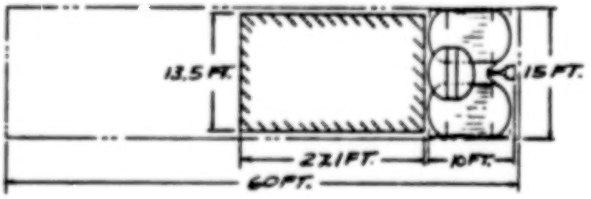
LSS SYSTEMS STOWED IN SHUTTLE BAY



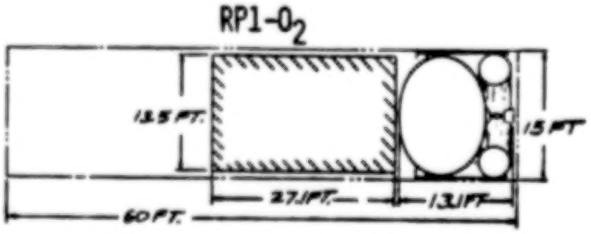
MODIFIED CENTAUR (RL-10, TANK IDLE MODE)



ADVANCED H_2-O_2



RP1-O_2



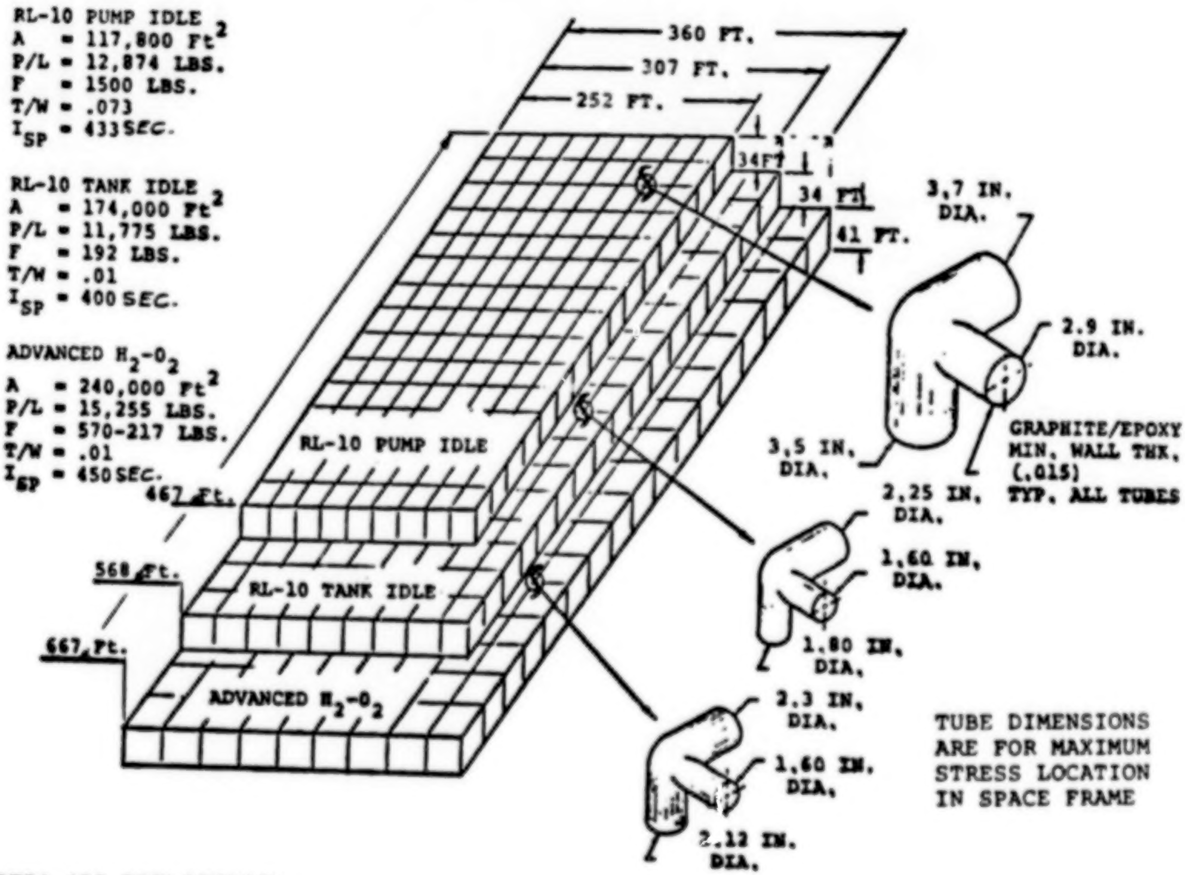
ADVANCED H_2-O_2 (SAME LSS AS RP1-O_2)

LSS PAYLOAD CAPABILITY

A comparison of deployed large space frame structures with specifications for their respective H_2-O_2 propulsion systems is shown in the next chart. The largest space frame (667 feet long by 360 feet wide by 41 feet deep) results from using the advanced H_2-O_2 (high specific impulse, low thrust) propulsion system. The smallest space frame shown results from using the Pratt and Whitney RL-10 engine in the pump idle mode. The associated high thrust-to-weight ratio (0.073) creates large stresses in the space frame members compared to a thrust-to-weight ratio of 0.01 for the other space frames in the chart. On the right hand side of the chart approximate space frame tube dimensions are shown for the maximum stress location in each tube. Graphite-epoxy tube materials were assumed with a minimum wall thickness of 0.015 inches.

It should be emphasized that the numbers in this chart (and throughout this paper) are preliminary. System and configuration optimization procedures have not been completed.

LSS PAYLOAD CAPABILITY


RL-10 PUMP IDLE
A = 117,800 Ft^2
P/L = 12,874 LBS.
F = 1500 LBS.
T/W = .073
I_{SP} = 433 SEC.
RL-10 TANK IDLE
A = 174,000 Ft^2
P/L = 11,775 LBS.
F = 192 LBS.
T/W = .01
I_{SP} = 400 SEC.
ADVANCED H_2-O_2
A = 240,000 Ft^2
P/L = 15,255 LBS.
F = 570-217 LBS.
T/W = .01
I_{SP} = 450 SEC.
360 FT.
307 FT.
252 FT.
34 FT
34 FT
41 FT.
467 Ft.
568 Ft.
667 Ft.
RL-10 PUMP IDLE
RL-10 TANK IDLE
ADVANCED H_2-O_2
3.7 IN. DIA.
2.9 IN. DIA.
3.5 IN. DIA.
GRAPHITE/EPOXY
MIN. WALL THK.
(.015)
TYP. ALL TUBES
2.25 IN. DIA.
1.60 IN. DIA.
1.80 IN. DIA.
2.3 IN. DIA.
1.60 IN. DIA.
2.12 IN. DIA.

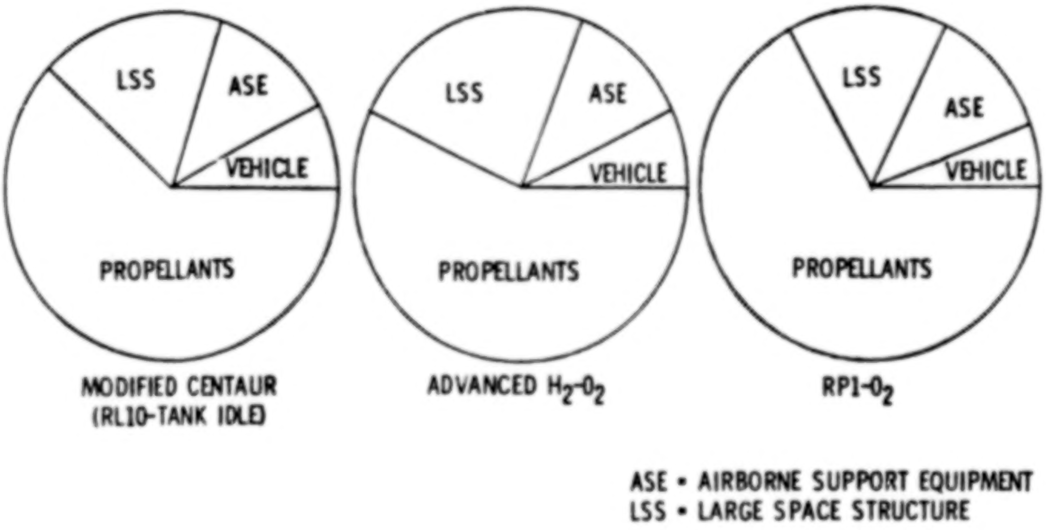
TUBE DIMENSIONS
ARE FOR MAXIMUM
STRESS LOCATION
IN SPACE FRAME

ALL NUMBERS ARE PRELIMINARY

WEIGHT DISTRIBUTIONS OF CONCEPTUAL DESIGNS FOR LARGE SPACE STRUCTURES SYSTEMS

The weight distributions of conceptual designs for large space structures systems are shown in pie charts in the next figure. Each pie represents a Shuttle cargo bay weight of 65,000 pounds. In each case the propellant fraction of the total weight is significantly greater than fifty percent. An airborne support equipment (ASE) weight of 8000 pounds was assumed for each case. Again, the heaviest (largest deployed area) payload results from using the advanced H_2-O_2 propulsion system. Note that the vehicle weight is not minimized by using the high specific impulse advanced H_2-O_2 engine. The RP1-O_2 vehicle weight is small because the RP1 fuel is much more dense than the H_2 fuel.

WEIGHT DISTRIBUTIONS OF CONCEPTUAL DESIGNS
FOR LARGE SPACE STRUCTURES SYSTEMS



CONCLUSIONS

- INTERACTIONS AMONG PROPULSION SYSTEM, PAYLOAD STRUCTURES AND SHUTTLE ARE IMPORTANT. FURTHER STUDY IS NEEDED.
- LOW THRUST-TO-WEIGHT RATIOS ARE DESIRABLE TO MAXIMIZE PAYLOAD WEIGHTS AND DEPLOYED AREAS.

PRIMARY PROPULSION/LARGE SPACE SYSTEM INTERACTIONS

Ralph H. Dergance
Martin Marietta Corporation

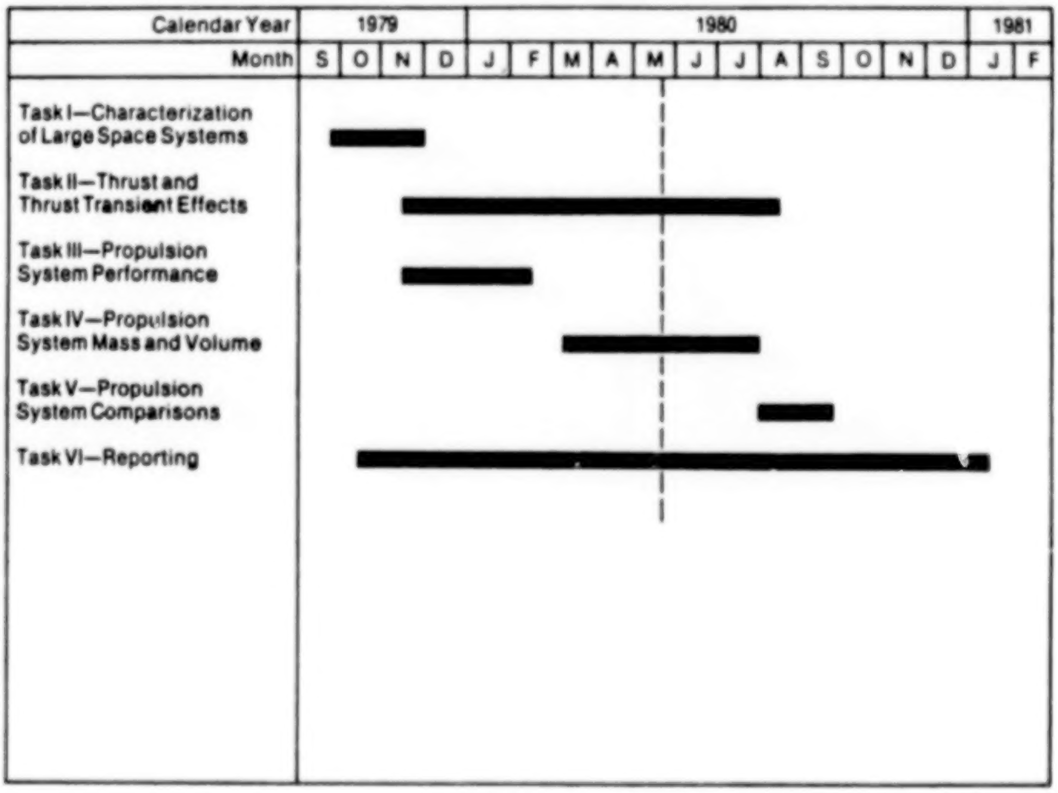
Contract Information

- Customer: NASA-Lewis Research Center
  Mr. Dean D. Scheer

- Contract Number: NAS3-21955

- Period of Performance: 20 September 1979 to 20 September 1980

Program Schedule



PP/LSSI Program Summary

The primary objective of the Primary Propulsion/Large Space System Interaction Study program is to determine the effects of low-thrust primary propulsion system thrust-to-mass ratio, thrust transients, and performance on the mass, area, and orbit transfer characteristics of large space systems.

PP/LSSI Task Objectives

Task I—Characterization of Large Space Systems—Determine the design characteristics of various classes of large space systems that are impacted by the primary propulsion thrust required to effect orbit transfer.

Task II—Thrust and Thrust Transient Effects—Determine the influence of primary propulsion steady-state and transient thrust on the mass and area of designated LSS concepts.

Task III—Propulsion System Performance—Determine the effect of selected primary propulsion system characteristics on deliverable payload mass from low earth orbit to high earth orbit.

Task IV—Propulsion System Mass and Volume—Determine the characteristics of selected pressure-fed and pump-fed stages for orbit transfer of LSSs and the effect of these stages and Space Shuttle constraints on mass and volume available for packaged large space systems.

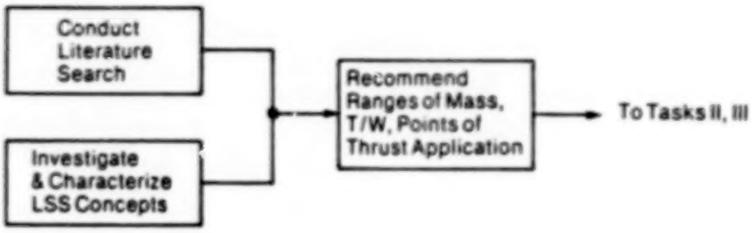
Task V—Propulsion System Comparisons—Determine relative merits of selected primary propulsion systems in terms of deliverable LSS mass, area, and/or length available for payload in the Orbiter cargo bay.

Task VI—Reporting—Monthly technical and financial reports, work plan, and program final report.

Task I - Characterization of Large Space Systems

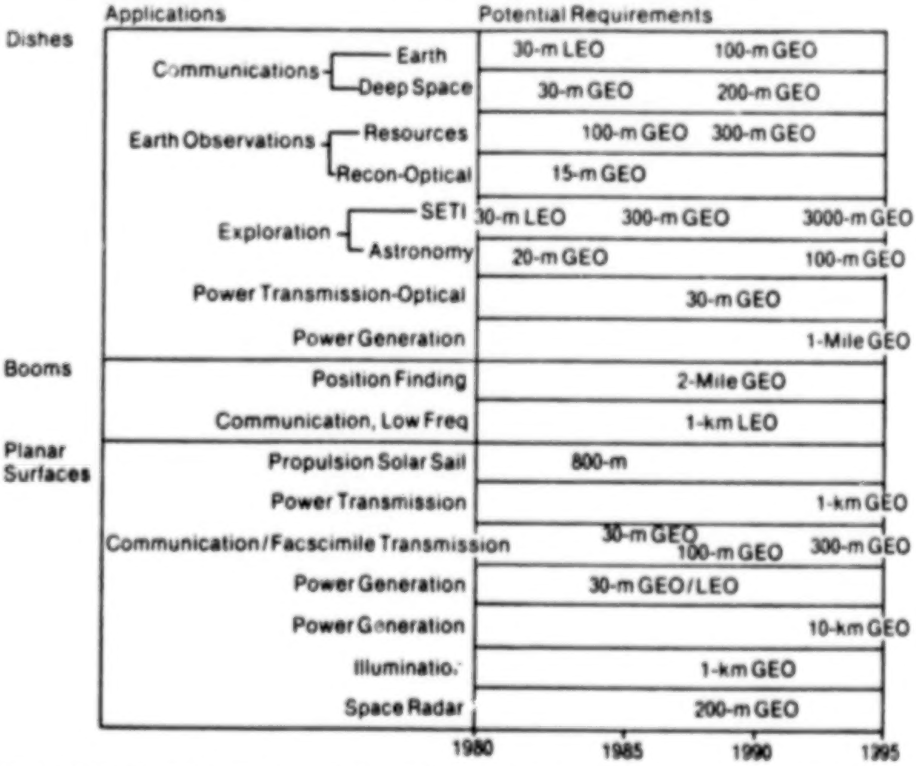
The goal of this task was to select 3 generic types of structural concepts and nonstructural surface densities that, when combined, would be representative of potential LSS applications.

Task I—Characterization of Large Space Systems


Conduct Literature Search
Investigate & Characterize LSS Concepts
Recommend Ranges of Mass, T/W, Points of Thrust Application
To Tasks II, III

- Identified and Evaluated More Than 120 References
- Investigated More Than 20 Potential LSS Missions & Concepts
- Categorized 14 LSS Concepts by Potential Usages
- Identified 4 Nonstructural Surface Densities Consistent with Missions

LSS Mission Parameters (Operational Altitude & Diameter)



Reference: "Toward Large Space Systems," *Astronautics and Aeronautics*, May 1977.

Structural Configurations

The following chart presents the 14 specific concepts that were investigated in Task I. The generic concepts to be evaluated in Task I - Thrust and Thrust Transient Effects - were selected from this population. Shown are design concept, the company responsible for the concept, and approximate diameter range compatible with a single STS mission.

Structural Configurations

- Umbrella Radial Rib Double-Mesh Antenna
  - Harris
  - 3 to 25 m
- Wrap Radial Rib Antenna
  - Lockheed
  - 30 to 300 m
- Erectable Radial Rib Antenna
  - General Dynamics
  - 30 to 200 m
- Radial Column Rib Antenna
  - Harris
  - 20 to 100 m
- Articulated Radial Rib Antenna
  - Harris
  - 20 to 40 m
- Maypole Antenna
  - Lockheed
  - 30 to 300 m
- Hoop & Column Antenna
  - Harris
  - 30 to 300 m
- Hoop & Column Radar
  - Grumman
  - 30 to 200 m
- Expandable Tetrahedral Truss Antenna
  - General Dynamics
  - 10 to 175 m
- Expandable Box Truss Antenna
  - Martin Marietta
  - 10 to 250 m
- Sunflower Solid Panel Antenna
  - TRW
  - 5 to 20 m
- Expandable Astrocell Module
  - Astro Research/Langley
  - 5 to 100 m
- Electrostatic Membrane
  - GRC
  - 5 to 200 m
- Expandable Box Truss Platform
  - Martin Marietta
  - 5 to 100 m

Note: Diameter limitations refer to single Orbiter packaging.

LSS Mission Parameters (Surface Mass Density)

The values shown are selected to provide surface mass densities representative of potential LSS payloads. The mesh surface (0.05 Kg/m^2) is typical for deployable mesh-type low frequency antennae. The high frequency surface (3.42 Kg/m^2) is representative of aluminized honeycomb panels or lump loading of a platform of ≈ 275 Kg/node. The radar antenna and power generation values were selected to include these types of payload in the population.

Bt utilizing these nonstructural surface densities in conjunction with the applicable structural concepts shown later, the full spectrum of potential payloads will be evaluated (mass and area) as a function of applied acceleration level.

LSS Mission Parameters (Surface Mass Density)

- Low-Frequency Antenna (< 20 gHz)
  - —Mesh Surface (i.e., Gold Plated, Moly Wire, Tricot Knit)
  - —Density = 0.05 kg/m^2 (0.01 lb/ft^2)

- High-Frequency Antenna (> 20 gHz)
  - —Rigid Panels (i.e., Aluminized Honeycomb Panels)
  - —Density = 3.42 kg/m^2 (0.70 lb/ft^2)

- Radar Antenna (1-2 gHz)
  - —Phased Array (3-Layer Lens)
  - —Density = 0.15 kg/m^2 (0.03 lb/ft^2)

- Power Generation
  - —Solar Cell Collector
  - —Density = 0.40 kg/m^2 (0.08 lb/ft^2)

Recommended Mission Parameters

The data presented below are in values selected for further evaluation in Task II. The diameter range (20-300 M) is compatible with the candidate concepts and nonstructural surface densities when constrained to launch a single payload in the cargo bay (allowances made for delivery stage volume). The surface mass densities were discussed on the preceding page.

The structural configurations selected are representative of tubular systems (Wrap Radial Rib), trusses and platforms (Expandable Box Truss), and a hoop and column (Grumman/Harris concepts).

Recommended Mission Parameters

- Diameter Range*:
  - —20 to 300 m
- Surface Mass Density
  - —0.05 to 3.42 kg/m^2
- Structural Configurations
  - —Wrap Radial Rib
  - —Hoop & Column
  - —Expandable Box Truss

*Actual diameter limitation based on packaging in Orbiter and payload limitations.

Recommended Configuration - Expandable Box Truss

The next 2 charts present the characteristics for the Expandable Box Truss. The diameter range is, again, approximate relative to cargo bay capability combined with the surface density range, which is representative of all potential payloads. The full range of surface densities is applied due to the truss' inherent load carrying capability. Representative missions are noted.

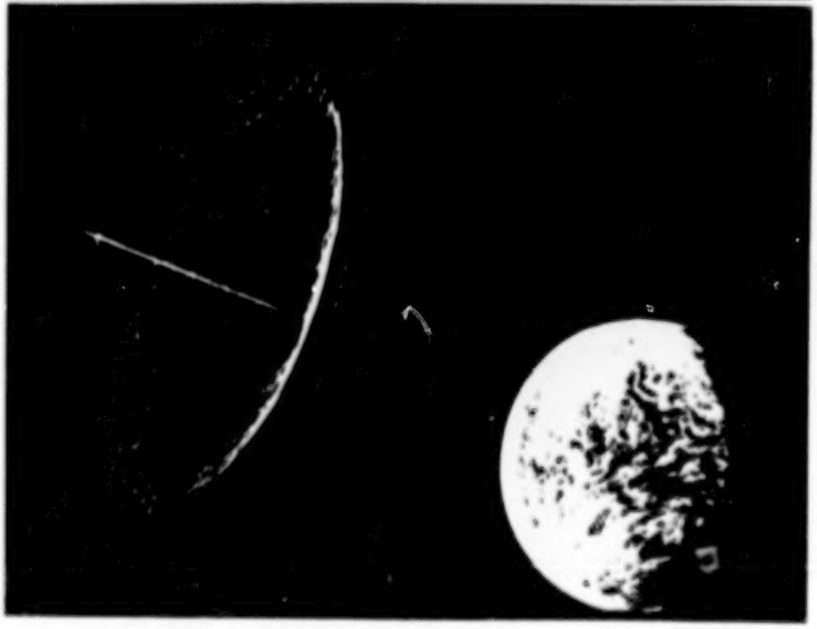
The point of thrust application to be used in the interaction analyses is at the center of the structure normal to its plane. These analyses will be first conducted with a single point of thrust application. Additional work will include multiple points that are yet to be determined.

The range of Thrust-to-Mass ratio to be evaluated is 0.02 to 1.0 g.

Recommended Configuration—Expandable Box Truss

- High-Frequency (< 20 gHz) Large-Diameter Reflector, Radar or Power Generator
  - —30 to 200-m Diameter
  - —0.05 - 0.15 - 0.40 - 3.42 kg/m^2
- Missions
  - —Communications
  - —Earth Observations
  - —Space Exploration
  - —Radar
  - —Power Generation
- Point of Thrust Application at Center of Structure Normal to Plane
- Thrust/Mass = 0.02 - 1.0 g

Expandable Box Truss Concept



Recommended Configuration - Hoop & Column

Data similar to those presented for the Expandable Box Truss are shown for the Hoop & Column Concept. The surface densities do not include the value associated with rigid panels since the Hoop & Column LSS concept is not compatible with deployment of these types of surfaces.

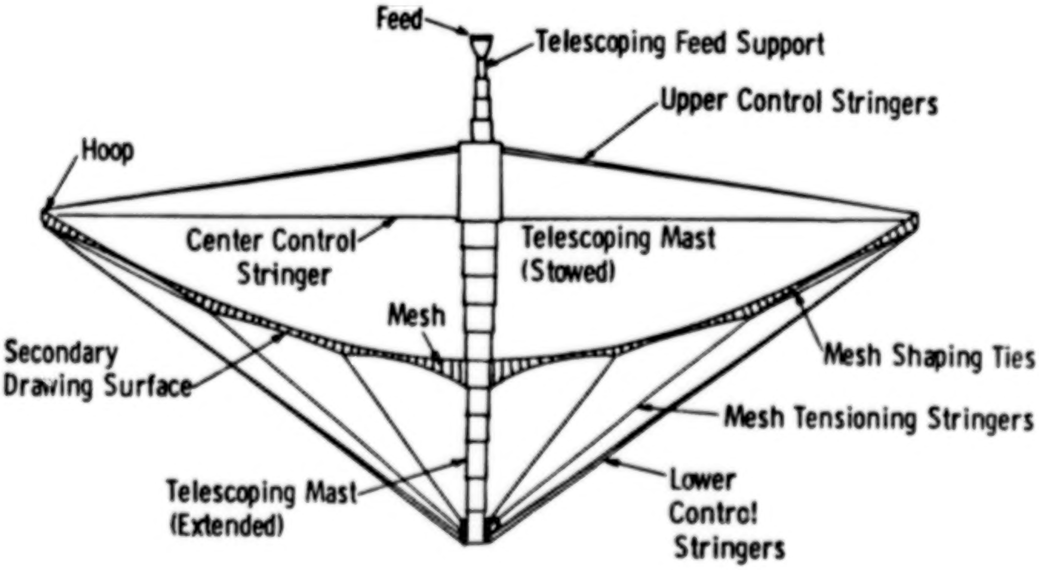
Again, representative missions are shown; the point of thrust application is at the end of the aft telescoping mast; and the Thrust-to-Mass ratio range is 0.01 to 1.0g.

Recommended Configuration - Hoop & Column

- Low-Frequency (< 20 gHz) Large-Diameter Reflector, Radar or Power Generation
  - 30 to 300-m Diameter
  - 0.05 - 0.15 - 0.40 kg/m²
- Missions
  - Earth Observations
  - Communications
  - Space Exploration
  - Radar
  - Power Generation
- Point of Thrust Application at End of Aft Telescoping Mast
- Thrust/Mass = 0.01 - 1.0 g*

*Structure probably limited to less than 1.0 g.

Hoop/Column Concept


Feed
Telescoping Feed Support
Upper Control Stringers
Hoop
Center Control Stringer
Telescoping Mast (Stowed)
Mesh
Secondary Drawing Surface
Mesh Shaping Ties
Mesh Tensioning Stringers
Telescoping Mast (Extended)
Lower Control Stringers

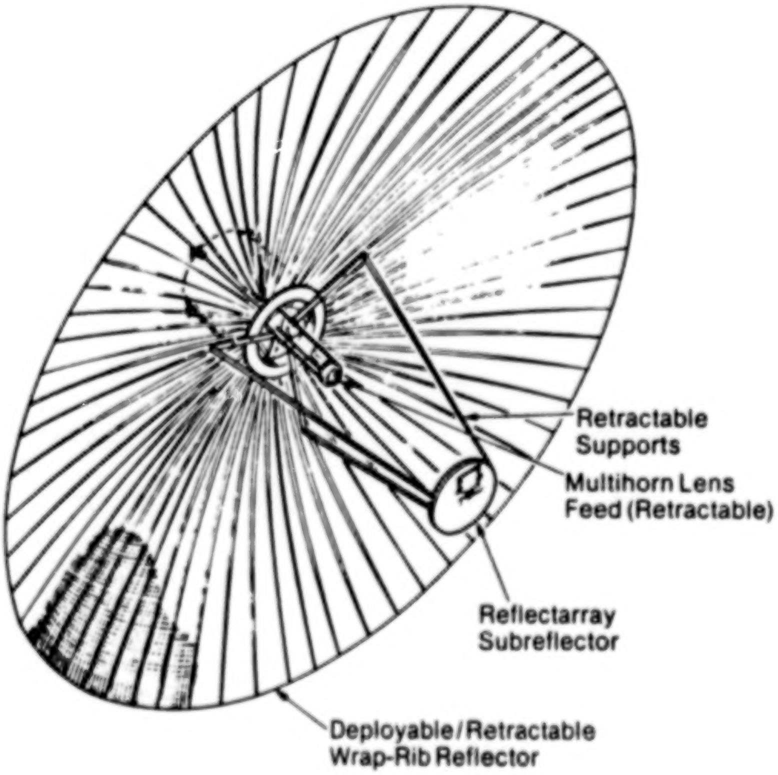
Recommended Configuration - Wrap Radial Rib

Similar data are presented for the Wrap Radial Rib concept. For this configuration, only mesh-type surfaces are considered (0.05 Kg/m^2) since the Wrap Radial Rib can only deploy this type of low frequency antenna.

Recommended Configuration—Wrap Radial Rib

- Low-Frequency (< 20 gHz) Large-Diameter Reflector
  - —30 to 300-m Diameter
  - —0.05 kg/m^2 Surface Density
- Missions
  - —Earth & Obervations
  - —Communications
  - —Space Exploration
- Point of Thrust Application at Hub
- Thrust/Mass = 0.02 - 1.0 g

Typical Lockheed Wrap-Rib Antenna: Deployed Configuration


Retractable Supports
Multihorn Lens Feed (Retractable)
Reflectarray Subreflector
Deployable/Retractable Wrap-Rib Reflector

Preliminary Diameter Limitations

To provide a realistic diameter range over which parametric mass and area relationships as a function of acceleration for Task III will be derived, the maxima presented on the facing page were determined.

The LSS payload value of 5440 Kg was derived by subtracting inert spacecraft mass (1360 Kg) from total mass in GEO (6800 Kg). These data were based on results of trajectory analyses previously performed and are representative of typical values for a cryogenic stage (Isp = 450 sec) with a mass fraction of = 0.85 and T/W = 0.05 g.

By combining the surface density with a structure with a total payload structure to nonstructure mass ratio of 1.5 and the maximum mass of 5440 Kg, the diameters shown result.

These values are only approximations but do bracket the range for the interaction analyses.

Preliminary Diameter Limitations

Surface Mass, kg/m²	Surface and Structure, kg/m²	Maximum Diameter, m
0.05	0.125	235
0.15	0.375	136
0.40	1.00	83
3.42	8.55	28

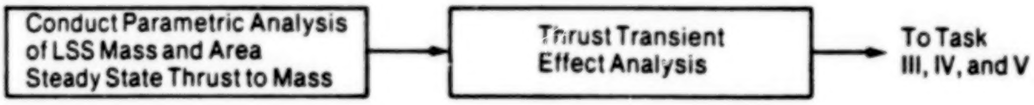
Note:
1. Typical payload ≈ 6800 kg (15,000 lb).
2. Typical Assumed Spacecraft ≈ 1360 kg (3000 lb).
3. Therefore, LSS payload ≈ 5440 kg (12,000 lb).
4. Typical low thrust-to-weight, structure/nonstructure ≈ 1.5.
5. Single Orbiter flight.

Task II - Thrust and Thrust Transient Effects

The principal output of this task will be LSS concept mass and area as a function of acceleration level during transfer from LEO to GEO. The analysis is divided in two parts - steady state and transient.

The key to the steady state analysis is starting with a representative minimum gage structural system. The criteria for minimum gage for the 3 structural concepts are shown. The iterative, rigorous finite element analysis is predicated upon failure of the structure when compared to failure modes such as Euler column buckling, local crippling, exceeding material allowables, etc. If any of these criteria are not met, the members are resized and the analysis is repeated.

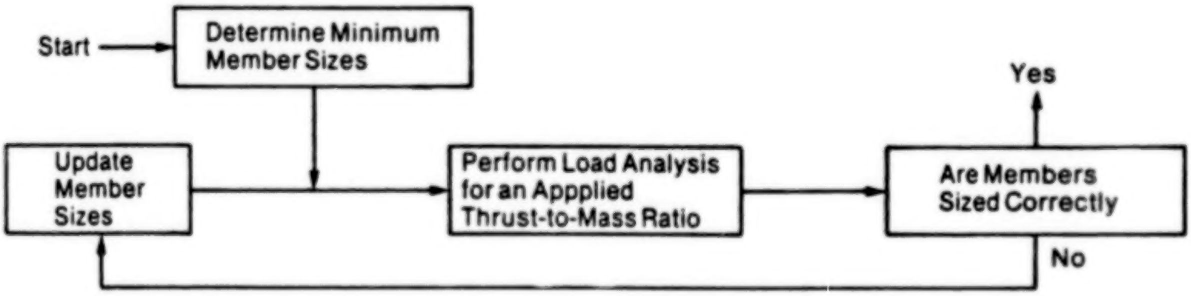
Task II—Thrust and Thrust Transient Effects


Conduct Parametric Analysis of LSS Mass and Area Steady State Thrust to Mass

Thrust Transient Effect Analysis

To Task III, IV, and V

- **Steady-State Analysis**


Start

Determine Minimum Member Sizes

Update Member Sizes

Perform Load Analysis for an Appplied Thrust-to-Mass Ratio

Are Members Sized Correctly

Yes

No

Minimum mass systems derived based on the following criteria:

- Expandable Box Truss—No member smaller than 3.8 cm (1.5 in.) diameter by 0.044 cm (0.0175 in.) thickness;
- Wrap Radial Rib—A baseline tapered rib for a 100-m-diameter design is scaled to maintain a tip deflection proportional to the antenna diameter under constant mesh loads;
- Hoop and Column—A maximum diameter hoop member at minimum gage is assumed, stay tapes are 2.5 cm (1.0 in.) by 0.044 cm (0.0175 in.), column based on Grumman-type design mass.

Task II - Thrust and Thrust Transient Effects (Concluded)

The transient analyses will evaluate the effects on mass and area of the structural concepts for two modes:

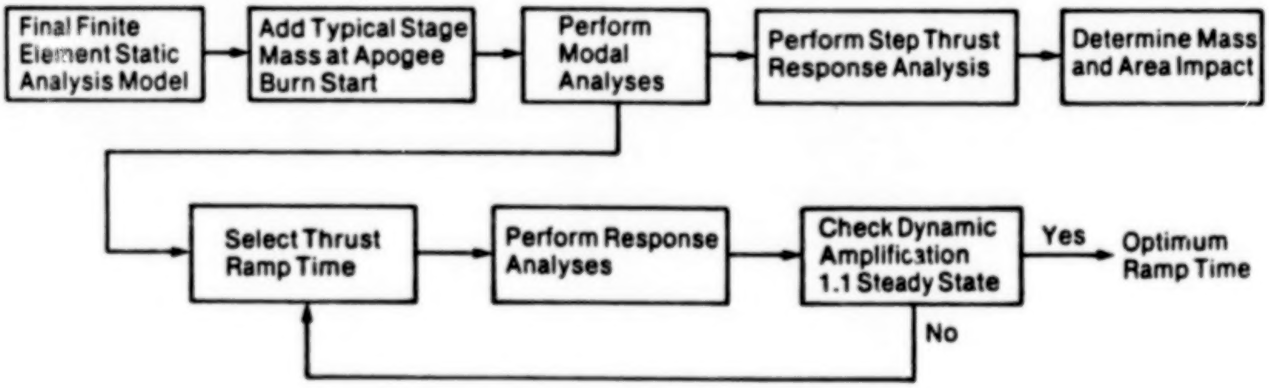
o A step input.

o A linear ramp input, varied to the point where dynamic amplification is $\leq$ 1.1 of the steady state value.

The 1.1 factor was selected to account for the effects of a multimode system when performing single mode system analyses. The value appears to be acceptable from a structural standpoint and achievable from an amplification standpoint.

Task II—Thrust and Thrust Transient Effects (concl)

- **Thrust Transient Effects Analysis**



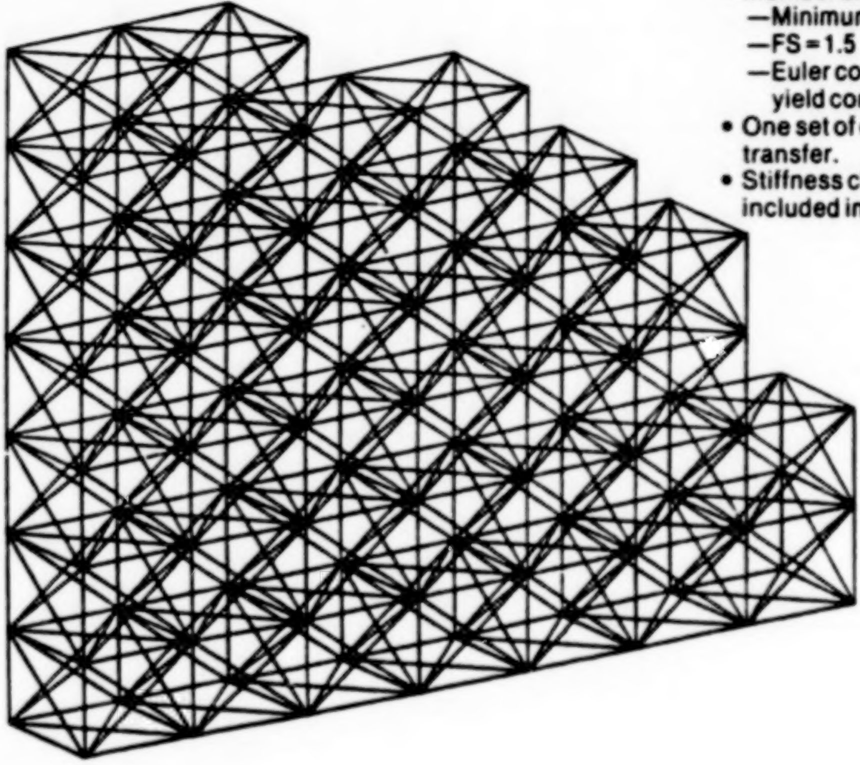
Final Finite Element Static Analysis Model
Add Typical Stage Mass at Apogee Burn Start
Perform Modal Analyses
Perform Step Thrust Response Analysis
Determine Mass and Area Impact
Select Thrust Ramp Time
Perform Response Analyses
Check Dynamic Amplification 1.1 Steady State
Yes
Optimum Ramp Time
No

- This analysis will be performed on representative configurations for 3 LSS concepts.
- Results will be extrapolated for remainder of configurations based on fundamental natural frequencies ($T_{ramp} = 1/f_n$).

Steady-State Structural Analysis Approach

Assumptions for Array Size Determination

- Structural and nonstructural masses lumped at nodes.
- Member weight determined using 20% margin.
- Inertial loads (maximum g) applied to nodes.
- Symmetric load condition.
- Member allowables determined using:
  - —Minimum Properties
  - —FS = 1.5
  - —Euler column, local bucking, and material yield considered as failure modes.
- One set of diagonals goes slack during orbit transfer.
- Stiffness characteristics of slack diagonals not included in finite element model.



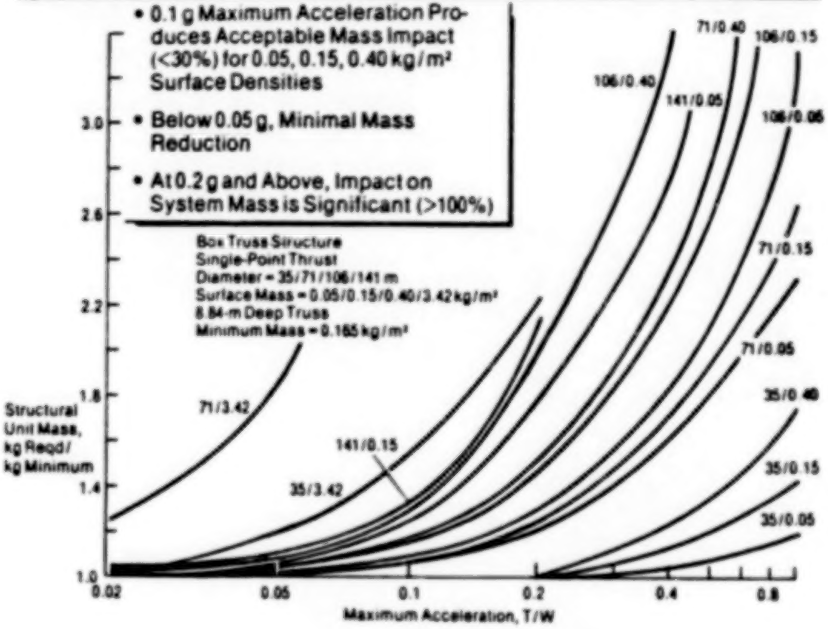
Expandable Box Truss - Unit Mass vs Thrust to Weight

This chart presents the results of the steady state analyses for the Expandable Box Truss. The structural unit mass is a factor of required mass to withstand the load applied divided by the minimum mass as represented by the previously presented minimum gage system.

All assumptions and conclusions are shown on the figure. It is interesting to note that 0.05 g is equivalent to 500 to 1000 lb_f of thrust, depending upon orbit transfer strategy, specific impulse and resultant payload weight. This thrust range appears to be best-suited for all diameters and surface densities except large (71 m) diameters with 3.42 Kg/m^2 nonstructural surface loading.

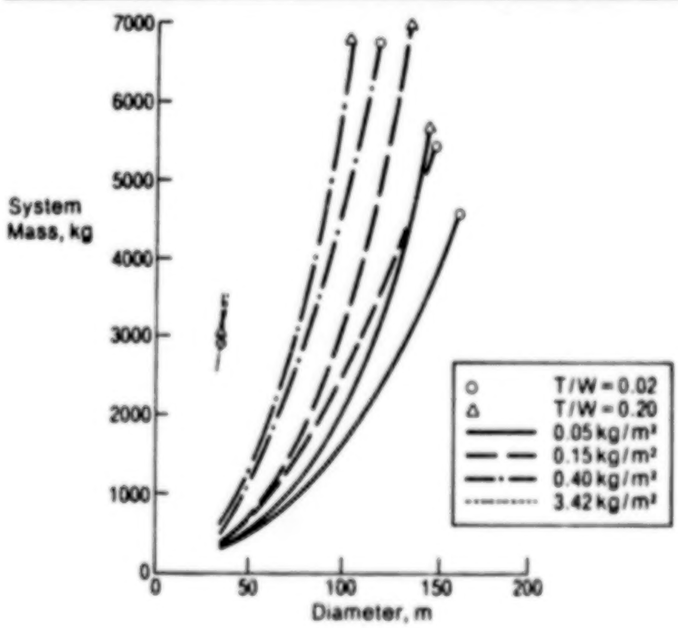
The structural weights include an allowance for joints, hinges, fittings, and diagonals. The baseline for these elements is again minimum gage and they increase in mass proportionally with the truss members.

Expandable Box Truss—Unit Mass vs Thrust to Weight


- 0.1 g Maximum Acceleration Produces Acceptable Mass Impact (<30%) for 0.05, 0.15, 0.40 kg/m^2 Surface Densities
- Below 0.05 g, Minimal Mass Reduction
- At 0.2 g and Above, Impact on System Mass is Significant (>100%)

Box Truss Structure
Single-Point Thrust
Diameter = 35/71/106/141 m
Surface Mass = 0.05/0.15/0.40/3.42 kg/m^2
8.84-m Deep Truss
Minimum Mass = 0.165 kg/m^2

Expandable Box Truss—System Mass vs Diameter



Structural Mass/Truss Depth Relationships

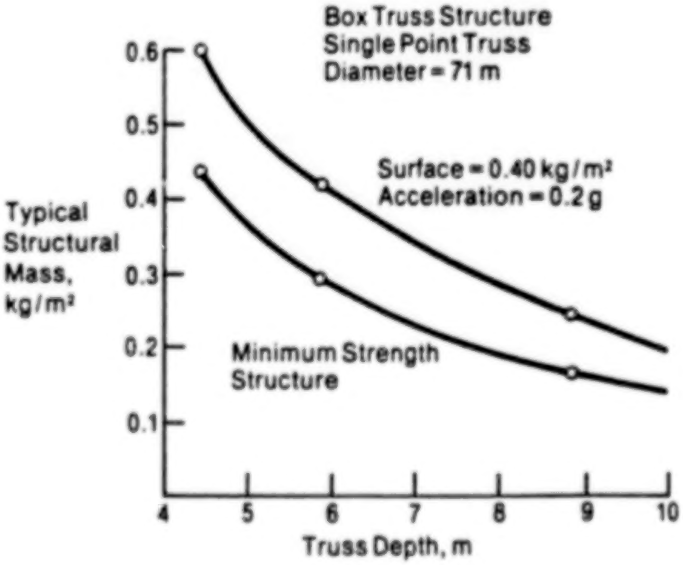
The graph presents data relating structural mass in Kg/m^2 to truss depth. The minimum strength curve shows the effect on mass as the truss depth decreases for an unloaded structure. The increase is caused by the necessity to add more fittings and mechanisms as the depth decreases for a fixed diameter array. For example, a 10 M truss cube is replaced by eight 5 M cubes with an attendant increase in corner fittings from 8 to 26.

The upper curve shows the effect of the surface ($0.40\ Kg/m^2$) on the structural mass. The divergence near the origin is attributed to the reduction of load - carrying capability of the truss as its depth decreases, resulting in an increase in individual member gage.

Since deeper trusses are inherently lighter and stronger, the conclusions that shorter transfer trusses are inherently lighter and stronger, with the single Orbiter flight constraint imposed in this study.

Structural Mass/Truss Depth Relationships

- Truss Structural Mass Decreases with Truss Depth because of Reduced Number of Fittings and Mechanisms
- Deeper Truss Also Reduces Impact of Orbit Transfer Load on Structural Mass
- Minimum Propulsion Stage Length Is Desirable to Maximize Truss Depth



Wrap Radial Rib Unit Mass vs Thrust to Weight

These data are similar to those presented on the Expandable Box Truss on page 120 herein.

The ground rules for sizing this concept are stated on the figure:

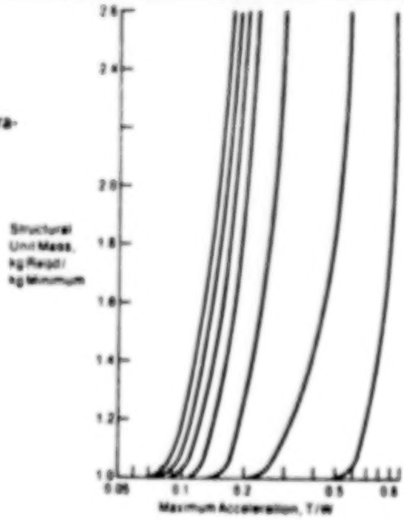
- Surface 0.05 Kg/m^2 (mesh antenna).
- Number of ribs proportional to $\sqrt{\text{diameter}}$.
- Rib deflection proportional to diameter - this is based on the premise that antenna performance is reduced as diameter increases and, therefore, deflection can increase with diameter.
- The baseline from which scaling was performed is a published 100 m, 96 rib Lockheed design.

In addition, a constant taper ratio (root to tip) of 3/1 was assumed. The baseline material of construction is graphite epoxy and the rib crossection was assumed to be elliptical with major to minor axis ratio of 5/1.

The results of this analysis indicate that accelerations between 0.05 and 0.10 g are preferred for the diameters considered. The diameters not shown are, from left to right, 194, 176, 158, 141, 106, 71, and 35 meters for the individual curves.

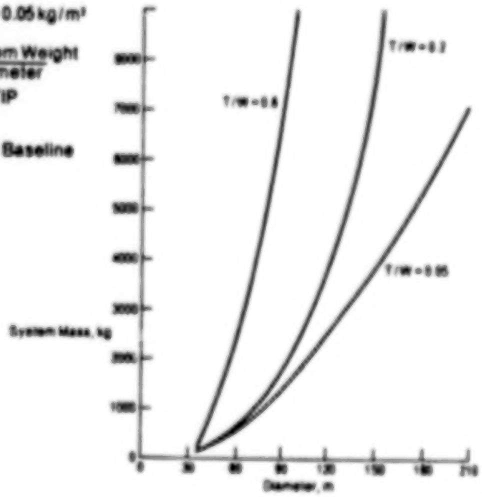
Wrap Radial Rib Unit Mass vs Thrust to Weight

- Baseline Parameters
  - —Surface = 0.05 kg/m^2
  - —Number Ribs $\propto \sqrt{\text{Diameter}}$
  - —Rib Deflection $\propto$ Diameter
  - —100-m Diameter
- Rib Highly Sensitive to Acceleration Level



Wrap Radial Rib—System Mass vs Diameter

- Surface Unit Weight = 0.05 kg/m^2
- Radial Rib Antenna
- Hub = 10% Total System Weight
- Number of Ribs $\propto \sqrt{\text{Diameter}}$
- Rib Size Defined by δ_{TIP} $\propto$ Diameter
- Graphite Epoxy Ribs
- 100-m Design Used as Baseline (96 Ribs)



Maximum Acceleration for <10% Structure Mass Impact

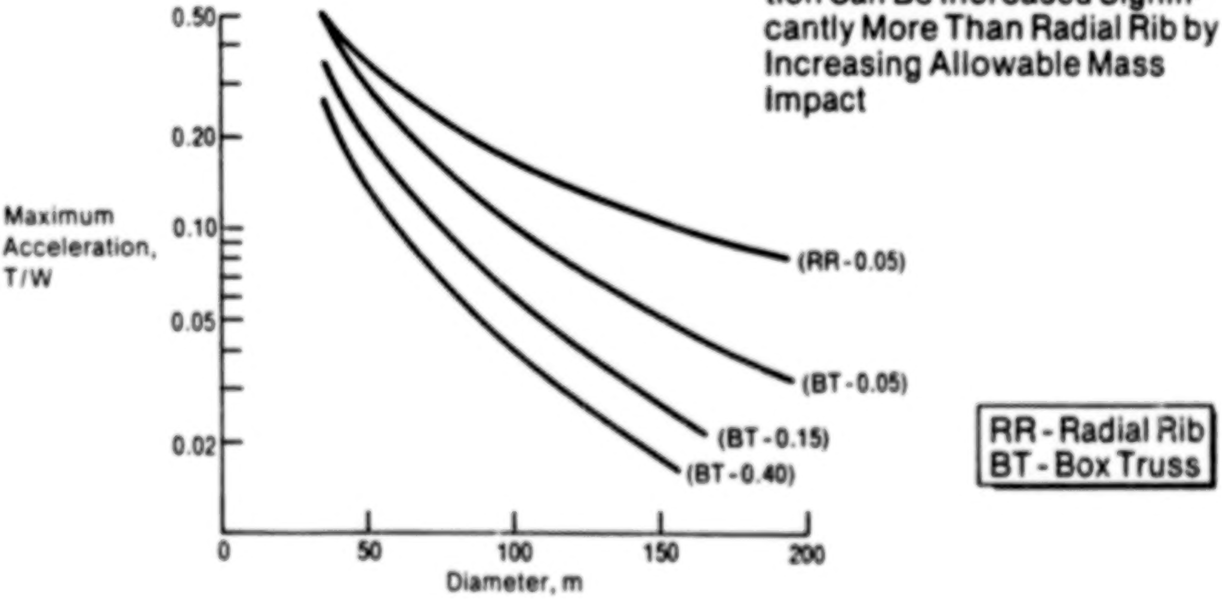
These curves present LSS diameter vs Acceleration level for the Wrap Radial Rib and Expandable Box Truss. Comparing the two concepts at a surface density of 0.05 Kg/m^2, it can be seen that the wrap rib has greater allowable acceleration capability than the box truss. This is primarily due to the stiffness of the ribs that results from the tip deflection constraint previously discussed.

The 10% mass impact was selected as a minimum. If this value is increased, the values for the truss and rib concepts will tend to converge due to the inherent load-carrying capability of the truss.

Acceleration levels between 0.05 and 0.10 g are again preferred for both concepts for diameters (150-200 m) compatible with a single Orbiter Flight.

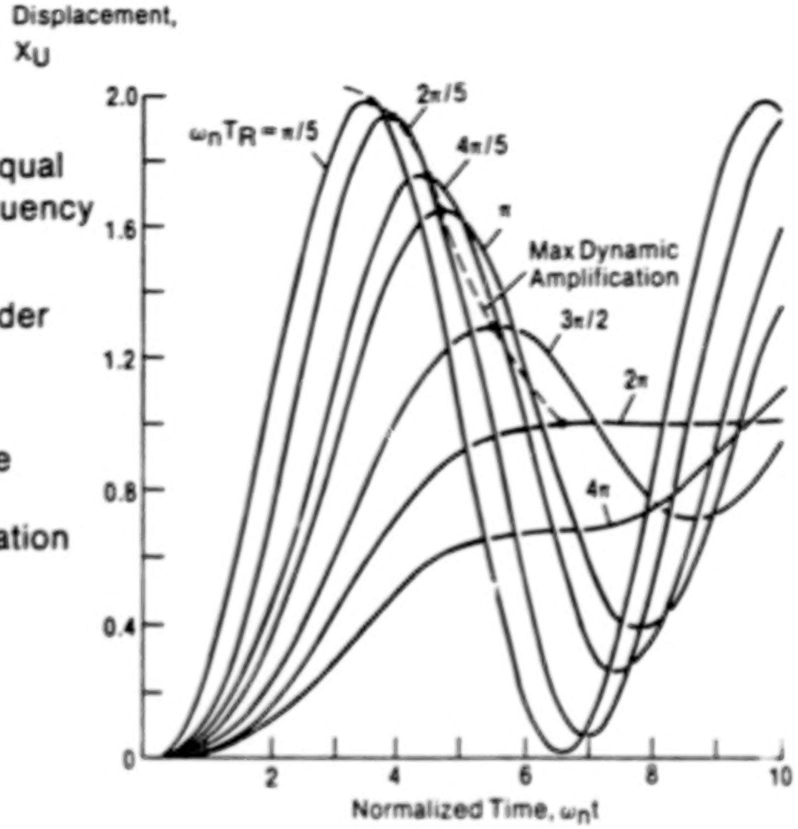
Maximum Acceleration for < 10% Structural Mass Impact

- Radial Rib Has Greater Allowable Acceleration at Large Diameters due to Stiffness Criteria That Increases Member Sizes with Diameter
- Box Truss Allowable Acceleration Can Be Increased Significantly More Than Radial Rib by Increasing Allowable Mass Impact



Response to Typical Ramp Input

- Optimum Ramp Time (T_R) equal to system fundamental frequency (f_n)
- For Real System, Higher Order Modes Modify This Pure Function
- Preliminary Results Indicate $T_R = 1/f_n$ Is Valid within 10% Amplification



Task III—Propulsion System Performance

Ground Rules

- Orbit transfer from LEO (160 n-mi circular orbit at 28.5° inclination) to GEO (19,368-n-mi circular orbit at 0° inclination)
- Initial Mass - 60,000 lbm
- Specific Impulse Range - 300 to 450 sec
- Number of Perigee Burns - 1 to 8
- Final Thrust-to-Mass Ratio Range - 0.01 to 1.0
- Constant Thrust and Constant Acceleration Analyses

Approach

- Three-Degree-of-Freedom Parameter Targeting and Optimization Program
- Thrust Segments Numerically Integrated
- Coast Segments Propagated using Keplerian Equations
- Gravity Turn during Perigee Burns
- Multiple Burns Split on Equal ΔV per Burn Basis
- Targeting Independent Variables
  - —Argument of Vehicle for Startup of Perigee Burns
  - —Apogee Altitude of Transfer Orbit
  - —Latitude of Startup of Apogee Maneuver
  - —Pitch and Yaw Attitude Angles during Final Orbit Insertion

Comparisons Between Constant Thrust and Constant Acceleration

The results of the trajectory analyses are summarized on the facing table and are presented graphically on pages 31 through 34 herein.

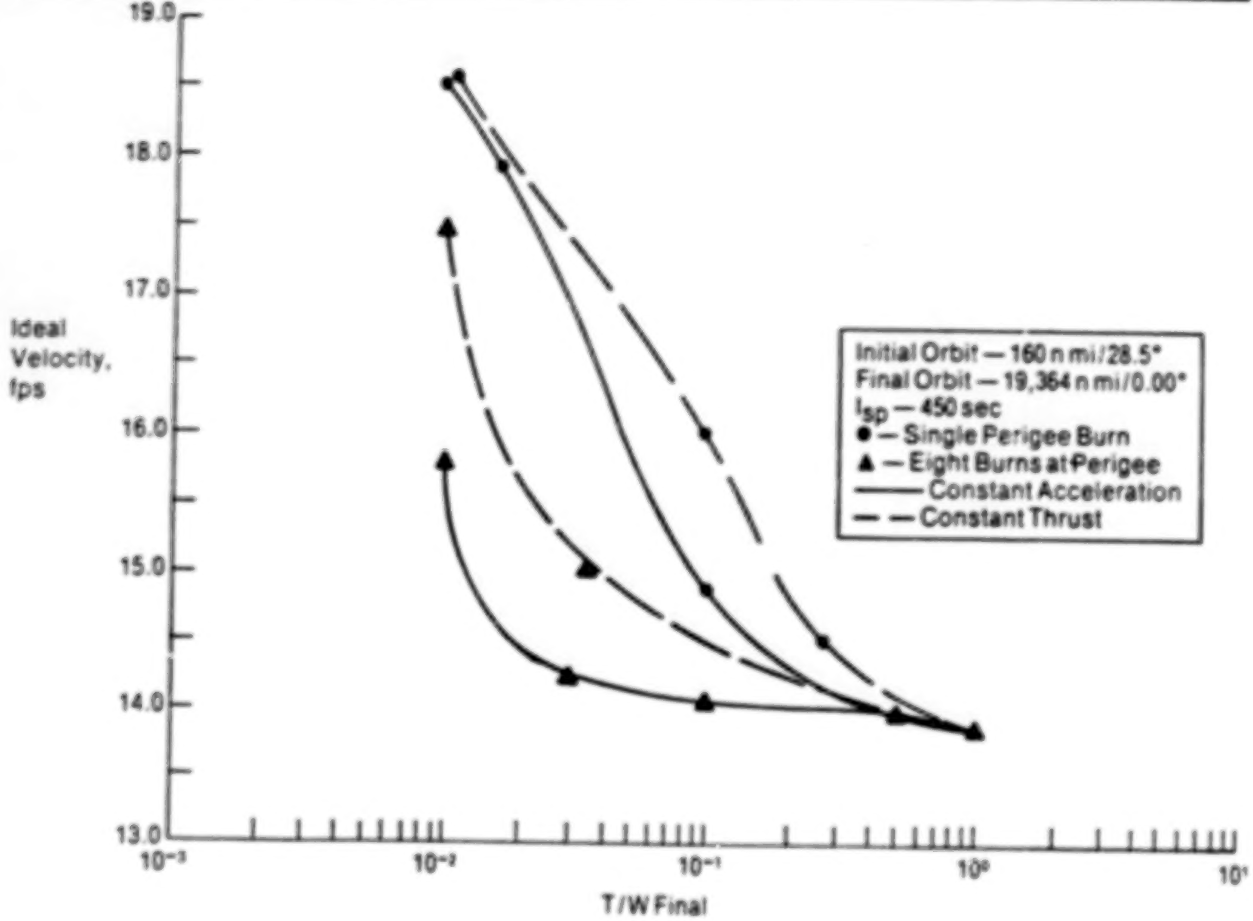
From these data, the following conclusions can be made:

- Constant acceleration (throttling) requires less ΔV than constant thrust.
- Constant acceleration requires less engine burntime than constant thrust.
- Constant acceleration produces shorter trip times than constant thrust.
- Constant acceleration results in increased payload capability when compared to constant thrust.
- 8 perigee burns are more efficient than a lesser number for all parameters except trip time where coast time dominates total mission duration.
- Acceleration between 0.05 and 0.10 g is preferred from a performance point of view and is compatible with the structure data previously discussed.

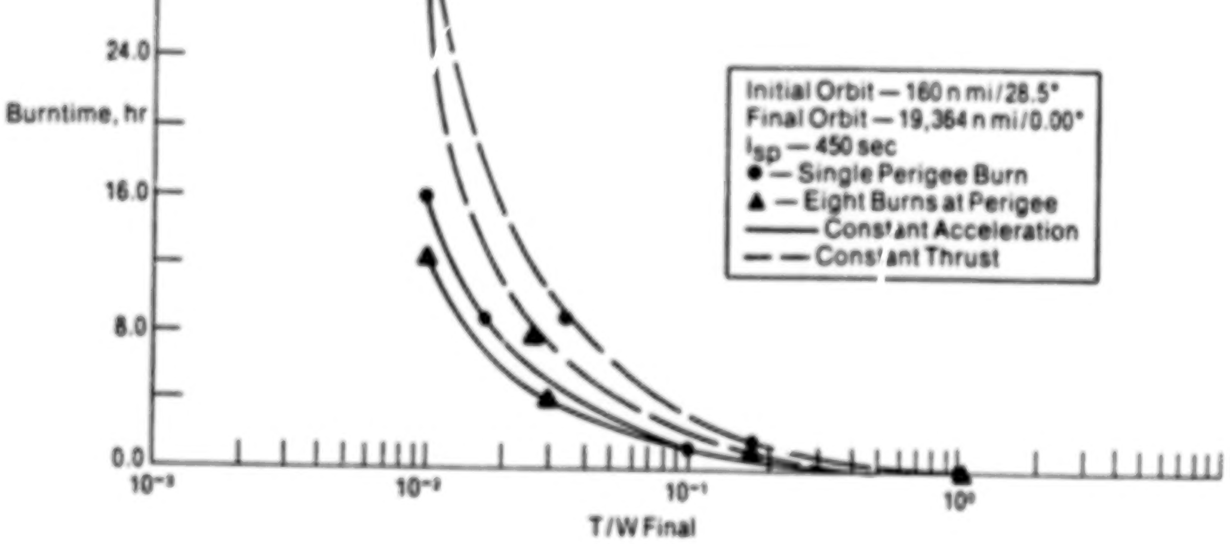
Comparisons between Constant Thrust and Constant Acceleration

Trajectory Variables	**Advantages/Disadvantages**
Velocity Requirement	• Constant thrust requires an 11% increase in ΔV over constant acceleration at low T/W. • Constant thrust requires a 2% increase at low T/W using one burn. • There is no significant difference in ΔV at T/Ws above 0.4. • ΔV transition occurs for both modes between 0.01 and 0.1 final T/W.
Burntime	• Small differences in total burntime between single and multiple burn transfers. • Constant thrust requires a 115% increase in burntimes relative to constant acceleration at low T/W.
Trip Time	• Constant thrust increases trip time by 65 to 88%, depending on the number of perigee burns. • Using high-thrust multiple burns, coast time dominates burntime; however, using low thrust, burntime dominates. • Multiple burn trip times are nearly invariant to T/W.
Payload	• Constant acceleration increases payload by 3 to 15% depending on the number of perigee burns employed. • There is no appreciable difference in payload performance above a T/W of 0.5.

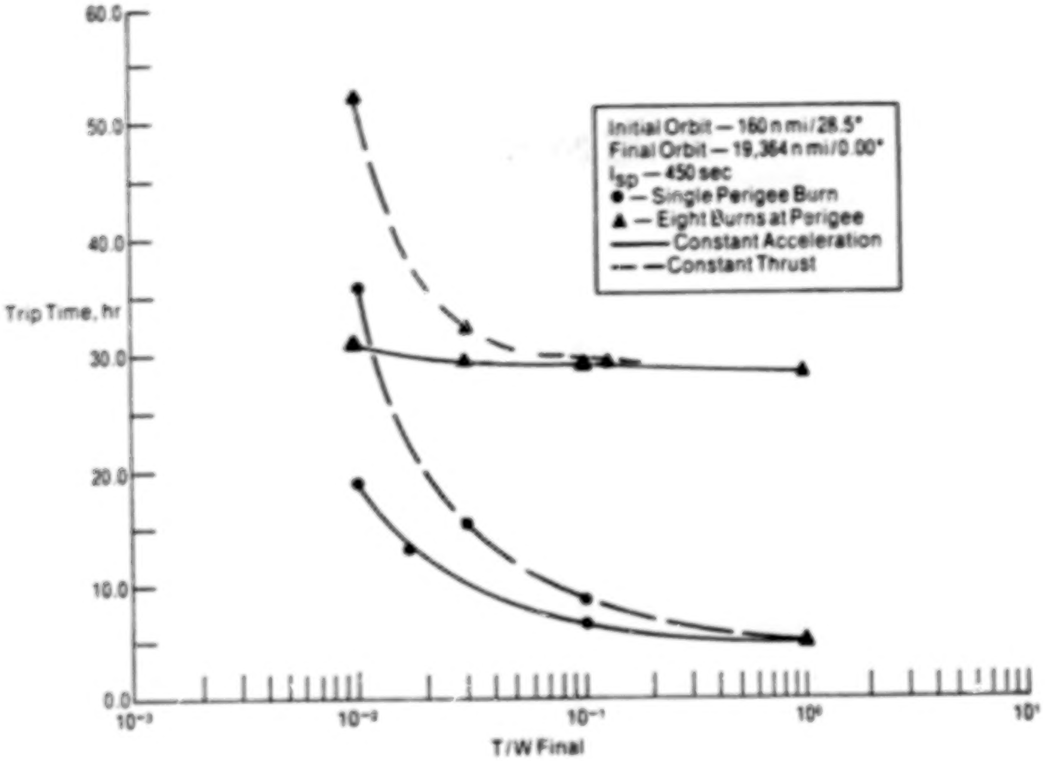
Ideal Velocity Requirements



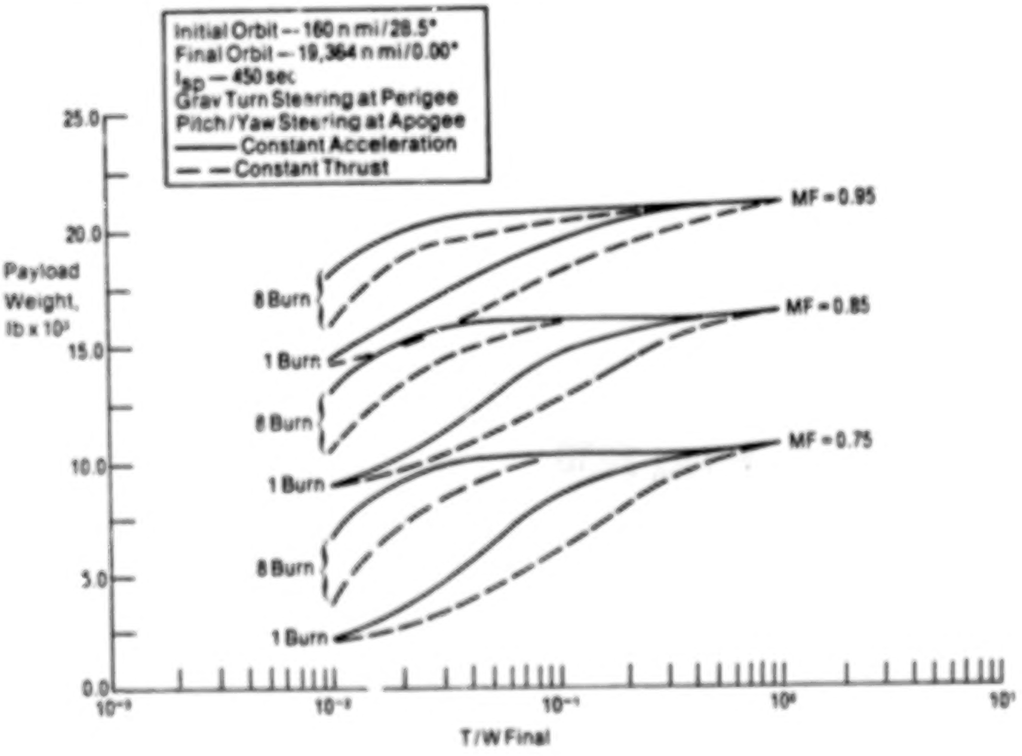
Burntime Requirements



Trip Time Requirements



Payload Capabilities vs T/W



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AUXILIARY CONTROL OF LSS

William Smith
Boeing Aerospace Company

The study objective is to provide a top level determination of auxiliary propulsion characteristics for broad classes of Large Space Structures. Boeing Aerospace Company under contract to NASA LeRC is conducting the investigation. The BAC study manager is J. P. Clark.

CONTRACT NAS3-21952

- PROJECT MANAGER: JOHN D. REGETZ, JR.
- PERIOD OF PERFORMANCE: 8/28/79 - 11/27/80
- 3350 MANHOURS

OBJECTIVE:

- DETERMINATION OF THE ELECTRICAL AND CHEMICAL PROPULSION CHARACTERISTICS AND TECHNOLOGY ADVANCES NECESSARY TO MEET AUXILIARY PROPULSION SYSTEM (APS) REQUIREMENTS ESTABLISHED FOR LARGE SPACE STRUCTURES (LSS)

TASKS

To accomplish the study objective we have broken the study into five major tasks. Generally, we determined LSS characteristics in Task 1, LSS disturbance forces and torques in Task 2, examined APS characteristics and requirements in Task 3, and will look at APS interactions with LSS in Task 4. Task 5 will be a comparison between the ideal APS characteristics and restrictions with currently available systems. This comparison should lead to the identification of specific technology advances needed in APS.

TASKS

1. CHARACTERIZATION OF LARGE SPACE STRUCTURES
    - LITERATURE SEARCH
    - DETERMINE LSS CHARACTERISTICS

2. ESTABLISHMENT OF DISTURBANCE CHARACTERISTICS
    - LITERATURE SEARCH
    - ANALYSIS OF DISTURBANCES

3. ESTABLISHMENT OF APS CHARACTERISTICS AND REQUIREMENTS
    - ANALYSIS OF CONTROL FORCES
    - ESTABLISH APS CHARACTERISTICS
    - ANALYSIS OF APS CHARACTERISTICS SENSITIVITIES

4. INTERACTION BETWEEN APS CHARACTERISTICS AND LSS CHARACTERISTICS
    - ANALYSIS OF LSS SENSITIVITIES
    - OPTIMUM APS DETERMINATION

5. DETERMINATION OF ELECTRICAL AND CHEMICAL PROPULSION TECHNOLOGY ADVANCES REQUIRED

STATUS AND ACCOMPLISHMENTS

Tasks 1-3 have laid the groundwork for the remainder of the study. In these tasks we identified seven generic classes of LSS, identified and analyzed disturbance forces on LSS, and established APS characteristics and qualitative sensitivities.

In Task 1 a literature search was conducted which looked at over 200 sources of information dealing with LSS missions and/or structures. There was an emphasis in this task on identifying generic structure classes and characteristic parameter ranges for each class. We used seven identified classes and idealized them into simple geometric shapes which could be easily modelled. Scaling laws were generated which allowed the seven ideal structures to be continuously scaled as to size and mass properties over their respective size ranges.

Task 2 identified relevant sources of disturbances and compared their effect on LSS. Based on the relative effects and on the applicability of the disturbances to the scope of the study, we selected those sources to be used in the later tasks. Along with each source, a quantification philosophy and methodology was developed.

These disturbances were applied over the range of scaling parameters in Task 3 to generate control force and torque requirements. In this task we identified important APS characteristics and established an APS characteristic sensitivity matrix.

STATUS AND ACCOMPLISHMENTS

TASK 1 - 3 COMPLETED

- o TASK 1 ACCOMPLISHMENTS
  - o LSS CHARACTERISTICS DETERMINED
  - o SEVEN GENERIC CLASSES IDENTIFIED
  - o IDEAL STRUCTURES AND SCALING LAWS GENERATED

- o TASK 2 ACCOMPLISHMENTS
  - o SOURCES OF DISTURBANCE IDENTIFIED
  - o DISTURBANCES ANALYZED AND COMPARED
  - o SELECTED SOURCES AND METHODS TO BE APPLIED IN LATER TASKS

- o TASK 3 ACCOMPLISHMENTS
  - o CONTROL FORCE AND TORQUE REQUIREMENTS DETERMINED
  - o IDENTIFIED IMPORTANT APS CHARACTERISTICS
  - o ESTABLISHED APS CHARACTERISTIC SENSITIVITIES

CHARACTERISTICS EXAMINED

The LSS characteristics looked at in Task 1 are summarized here. The mass properties included total mass, mass distribution and inertias. Orientation requirements were defined by pointing accuracy and slew requirements. Area distribution included the location of radar panels, the solid surfaces, antennas and trusses. The orbit parameters were the range of altitudes and eccentricity needed and figure accuracy requirements were defined for each mission examined.

CHARACTERITICS EXAMINED

- MASS PROPERTIES
- ORIENTATION REQUIREMENTS
- AREA DISTRIBUTION
- ORBIT PARAMETERS
- FIGURE ACCURACY

CHARACTERIZATION OF LARGE SPACE STRUCTURES

This chart shows the breakdown on the generic classes into three main categories - planar structures, single antenna systems, and multiple antenna systems. To better fit the wide range of structures examined, we subdivided each of these classes into two or three subclasses. These subclasses are as follows:

1. Planar Structures
    - A. Large flat array
    - B. Cross structure

2. Single Antenna Systems
    - A. Box structue
    - B. Modular antenna system
    - C. Maypole or hoop and column antenna

3. Multiple Antenna Systems
    - A. Modular antenna farm
    - B. Multiple antenna farm

These structures are illustrated in the next three charts.

TASK 1 CHARACTERIZATION OF LARGE SPACE STRUCTURES

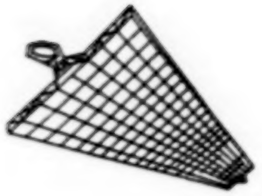
GENERAL CLASSES

1. PLANAR STRUCTURES
    - A. LARGE FLAT ARRAY
    - B. CROSS SHAPED STRUCTURES

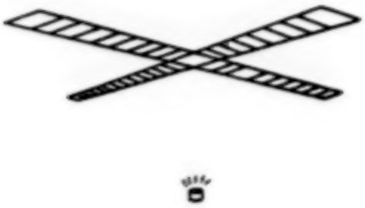
2. SINGLE ANTENNA SYSTEMS
    - A. BOX STRUCTURE
    - B. MODULAR STRUCTURE
    - C. MAYPOLE ANTENNA

3. MULTIPLE ANTENNA SYSTEMS
    - A. MODULAR ANTENNA FARM
    - B. MULTIPLE ANTENNA FARM

- PLANAR STRUCTURES

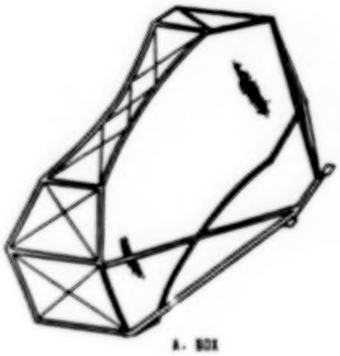


A. LARGE FLAT ARRAY

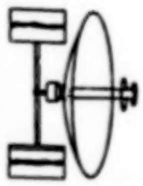


B. CROSS STRUCTURE

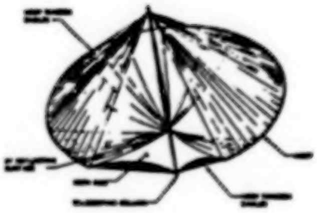
- SINGLE ANTENNA SYSTEMS



A. BOX

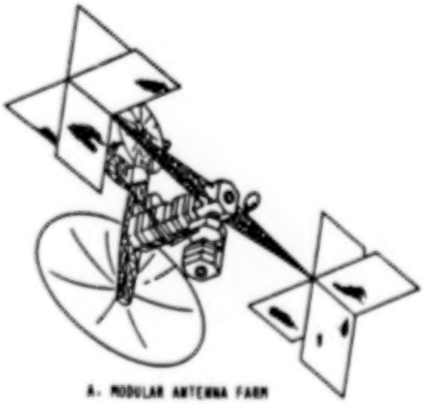


B. MODULAR

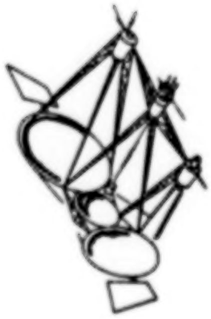


C. MAYPOLE

- MULTIPLE ANTENNA SYSTEMS



A. MODULAR ANTENNA FARM



B. MULTIPLE ANTENNA FARM

SCALING PARAMETER SELECTION

For each of the ideal classes, a single parameter was established from which all structures were scaled. This ideal scaling parameter was generally associated with area but took different form for each class. Listed here are the classes, the scaling parameter, the parameter range, and the corresponding mass range based on the scaling laws established.

SCALING PARAMETER SELECTION

CLASS		STRUCTURE	CHARACTERISTIC PARAMETER	PARAMETER RANGE	MASS RANGE (KG)
I	PLANAR	PLATE	LENGTH	30 - 21000 (M)	170 TO 8.27×10^7
		CROSS	LENGTH	40 - 4000 (M)	560 TO 56000
II	SINGLE ANTENNAS	BOX	LENGTH	82 - 1300 (M)	1.23×10^5 TO 1.95×10^5
		MODULAR ANTENNA	ANTENNA DIA	15 - 200 (M)	2050 TO 27000
		MAYPOLE	ANTENNA DIA	30 - 1500 (M)	100 TO 2640
III	MULTIPLE ANTENNAS	ANTENNA FARM	ANTENNA DIA	15 - 60 (M)	3000 TO 12000
		SERIES OF ANTENNAS	NUMBER OF ANTENNAS	2 - 10	44000 TO 216500

DISTURBANCE CLASSIFICATION

To accommodate the wide range of altitudes and eccentricity requirements, we groundruled four disturbance classifications. The assumption implicit is that the structure will be erected/deployed in LEO, transferred to GEO while providing thrust vector control through slewing of the vehicle with the LSS auxiliary propulsion, and finally stationkept at GEO. One must look at the maximum disturbances at both LEO and GEO to size the system for a worst case distrubance. However, because of the wide separation of requirements in a maximum and nominal case, it was felt that nominal and maximum requirements should be analyzed separately and correspondingly, different APS systems defined.

DISTURBANCE CLASSIFICATIONS

- MAXIMUM DISTURBANCE AT LEO (300 KM)
  - WORST CASE ORIENTATION
- MAXIMUM CONTROL TORQUES DURING LEO-GEO TRANSFER
  - THRUST AXIS FOR EACH VEHICLE DETERMINED
- NOMINAL GEO ON-ORBIT REQUIREMENTS
  - NOMINAL ORIENTATION
- MAXIMUM DISTURBANCE AT GEOSYNCHRONOUS ORBIT
  - WORST CASE ORIENTATION

SUMMARY OF DISTURBANCES

In Task 2 we identified, analyzed and compared various disturbance forces and torques on LSS. Based on this process we selected those sources to be included in the study. We did not include magnetic and thermal disturbances.

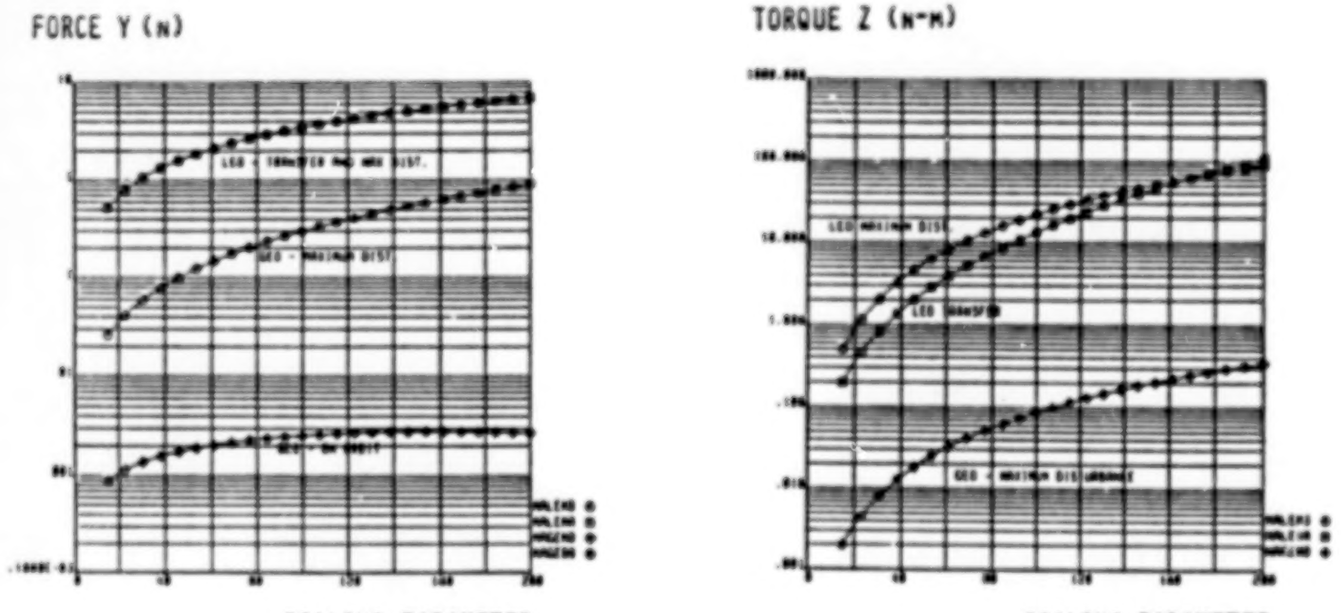
Magnetic disturbances are not likely to be significant unless large current loops are present in the vehicle. These loops are very mission dependant and were not considered relevant in our broad study. Likewise, thermal disturbances while clearly significant to LSS are both mission dependent and very difficult to analyze. Furthermore, it is unlikely that a thruster will be used to provide a restoring force for thermal disturbance.

DISTURBANCE	INCLUDED	COMMENT
RADIATION	YES	PHOTON PRESSURE, EARTH ILLUMINATION
GRAVITY GRADIENT	YES	MOST SIGNIFICANT DISTURBANCE
AERODYNAMIC	YES	INCLUDED BELOW 1000 KM
MAGNETIC	NO	DISTURBANCE RELATIVELY INSIGNIFICANT
THERMAL	NO	TOO MISSION DEPENDANT TO BE CONSIDERED
STATIONKEEPING	YES	INCLUDED AT GEOSYNC.

MODULAR SINGLE ANTENNA

The significant disturbance effects were evaluated at each condition for each generic LSS class and summed over the scaling parameter range. The result is a series of curves of the disturbance forces and torques. The chart shows two such plots, one for the force in the Y direction (normal to the orbit plant) and the other for torque about the Z axis (the local vertical). These illustrations are typical only but do show the wide range of effects that generally occur.

MODULAR SINGLE ANTENNA



SIGNIFICANT APS CHARACTERISTICS TO BE DETERMINED

The significant APS characteristics were identified by considering the basic control tasks of attitude control, shape control and stationkeeping.

Attitude control consists, ideally, of exact cancellation of disturbance torques. The ideal can be closely approximated by delivering periodic torque impulse bits. Thrust level and modulation are thus important characteristics. Transient effects such as the rise and decay profiles may also be significant if limit cycle operation is employed. The significant characteristics for attitude control are then thrust level, modulation and transient effects.

Shape control implies a distributed system thus the number and distribution of thrusters is an added significant characteristic.

Stationkeeping is not a demanding task in general and no additional characteristic appears important.

The four characteristics uncovered above - thrust level, number and distribution of thrusters, modulation and transient effects are operating characteristics. From a systems viewpoint the allowable APS mass must be considered and this has been added as a fifth significant characteristic.

SIGNIFICANT APS CHARACTERISTICS TO BE DETERMINED

- o NUMBER AND DISTRIBUTION OF THRUST UNITS
- o MAXIMUM-MINIMUM THRUST LEVELS
- o RISE AND DECAY PROFILES
- o THRUST MODULATION
- o ALLOWABLE MASS

SENSITIVITY MATRIX

The sensitivity matrix was developed by considering the possible interaction between each of the five identified significant APS characteristics and the major attitude control functions.

The number and distribution of thrusters are particularly important in a shape control application. For more rigid structures the effects are of little consequence. Thrust level is significant in most attitude control functions. It is omitted from the shape control column because timing is more important than thrust level for active damping. Rise and decay characteristics affect the timing of thrust pulses so this too is significant for shape control. Transients also influence limit cycle performance and thus pointing accuracy. Modulation and allowable mass interact widely with most of the attitude control functions.

SENSITIVITY MATRIX

ATTITUDE CONTROL FUNCTIONS / APS CHARACTERISTICS	ATTITUDE CONTROL			SHAPE CONTROL	STATION-KEEPING	DESAT-URATION
	DISTURB. CANCEL.	POINTING	MANEUVER			
NO. AND DISTRIBUTION				S		
THRUST LEVEL	S	S	S		S	S
RISE AND DECAY		S		S		
MODULATION	S	S	S	S	S	S
ALLOWABLE MASS	S	S	S		S	S

FUTURE WORK

Tasks 1 through 3 in many respects lay the groundwork for the remaining work. First the interaction between APS and LSS are to be determined. This task is in effect the description of the parameter relationships; i.e., the process of quantifying the qualitative sensitivities identified in the previous chart. Once this has been accomplished it will be possible to define the ideal APS for control of LSS. Different characteristics may be desirable for difficult classes and there may be variations as a function of the scaling parameter.

In the final task, the desired characteristics will be compared with those available in state of the art and projected systems. Discrepancies will indicate areas in which APS technology advances would be profitable.

FUTURE WORK

TASK 4 INTERACTION BETWEEN APS CHARACTERISTICS AND LSS CHARACTERISTICS

4.1 ANALYSIS OF LSS SENSITIVITIES
- o EXTEND SENSITIVITY STUDIES TO INCLUDE EFFECT ON LSS OF
  - I_{sp}
  - MASS OF APS SUPPORTING EQUIPMENT (TANKS, PPU'S, POWER SUPPLY, ETC)
  - STRUCTURAL STIFFNESS

4.2 OPTIMUM APS DETERMINATION
  - DEFINE IDEAL CHARACTERISTICS FOR CONTROL OF LSS

TASK 5 DETERMINATION OF ELECTRICAL AND CHEMICAL PROPULSION TECHNOLOGY ADVANCES REQUIRED

- o COMPARE EXISTING CHARACTERISTICS AND CAPABILITY WITH THOSE DESIRED TO IDENTIFY DEFICIENCIES

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EFFECT OF ORBITAL TRANSFER LOADS ON LARGE PLATFORMS

Joseph E. Walz, Harold G. Bush,
Walter L. Heard, Jr., and John J. Rehder
NASA Langley Research Center

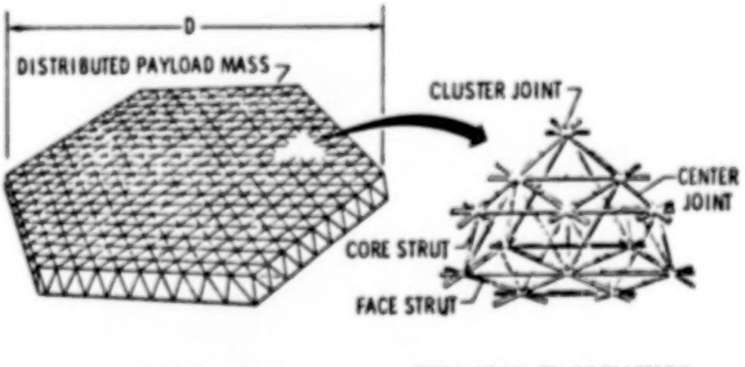
SPACECRAFT CONFIGURATION

(Figure 1)

The general outline of this presentation is to first discuss a preliminary automated structural sizing procedure suitable for conceptual design and early tradeoff studies of large truss platforms configured for Shuttle transportation to LEO. Then some orbital transfer design considerations are discussed. Finally, platforms that are sized to withstand orbital transfer loads for the LEO to GEO maneuver are compared to platforms sized only for LEO application.

The first figure depicts a flat tetrahedral truss of hexagonal planform. The maximum dimension of the platform is designated as D. There is a uniformly distributed functional surface attached to one face of the platform. This nonstructural surface is termed the payload mass, M_p. The top face of the platform can be thought of as composed of "rings". The number of rings can be identified by the number of members along an edge of the top surface. The blowup of a small portion of the truss indicates that the top and bottom surfaces are constructed of face columns or struts. The top and bottom surfaces are separated by core struts, and all struts are interconnected by cluster points which accomodate nine struts per node. The face struts contain a hinged center joint to permit packaging.

SPACECRAFT CONFIGURATION



HEXAGONAL PLANFORM TETRAHEDRAL TRUSS PLATFORM

DEPLOYABLE PACKAGING

(Figure 2)

This presentation considers only deployable trusses although information on both deployable and erectable trusses is contained in references 1 and 2. The left side of the figure identifies the six sizing variables used in the optimization process, namely; the lengths, outer diameters, and thickness of face and core struts. All face struts are identical as are all core struts.

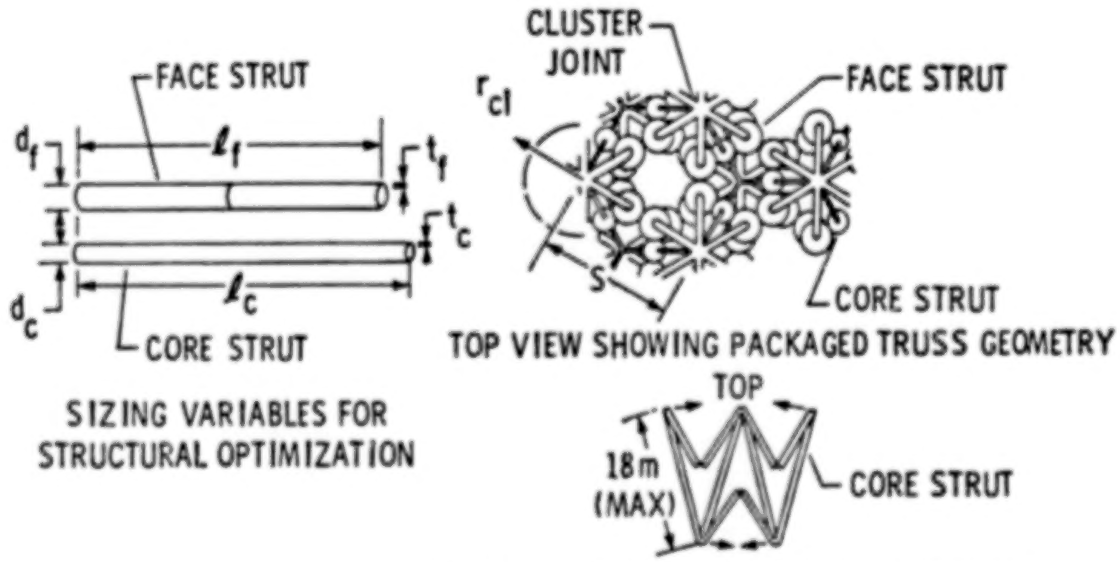
Both inward and outward folding trusses have been examined. In most instances the outward folding truss is the least efficient, therefore the results presented here are for the inward folding truss. Note that for the inward folding truss, the face strut length can be no greater than the core strut length for tight packaging, and the core strut length can be no greater than 18 m because of the cargo bay length of the Space Shuttle.

The upper right sketch depicts a planview of the platform in the tightest packaged configuration (structure only, with no surface covering material). In this view the axes of all struts are oriented perpendicular to the plane of the paper so that the struts appear as circles. The larger circles indicate face strut halves and the smaller circles indicate core struts.

REFERENCES

1. Heard, W. L., Jr.; Bush, H. G.; Walz, J. E.; and Rehder, J. J.: Structural Sizing Considerations for Large Space Platforms, AIAA Paper No. 80-0680, presented at the 21st Structures, Structural Dynamics and Materials Conference, May 12-14, 1980.
2. Bush, H. G..; Heard, W. L., Jr.; Walz, J. E.; and Rehder, J. J.: Deployable and Erectable Concepts for Large Spacecraft, SAWE Paper No. 1374, presented at the 39th Annual Conference of the Society of Allied Weight Engineers, Inc. May 12-14, 1980.

DEPLOYABLE PACKAGING



SIZING VARIABLES FOR STRUCTURAL OPTIMIZATION

TOP VIEW SHOWING PACKAGED TRUSS GEOMETRY

PACKAGE WITH INWARD FOLDING FACE STRUTS

STRUCTURAL OPTIMIZATION APPROACH

(Figure 3)

Several different math programming routines are available for optimization purposes. The one used for this study is CONMIN (ref. 3). The platform structural mass per unit area was minimized with respect to the sizing variables. Upper and lower bounds are used to constrain the sizing variables. The platform was required to have a natural frequency greater than or equal to a specified design value (i.e. to permit control). The individual struts were required to have a natural frequency which was a multiple of the platform design frequency to avoid coupling. The Euler buckling loads of the struts were required to be greater than or equal to the imposed loads. Loads due to deployment were assumed small since controlled deployment was assumed. Loads due to gravity gradient were considered but were found to be insignificant.

REFERENCE

3. Vanderplaats, Garret N.: CONMIN - A FORTRAN Program for Constrained Function Minimization. User's Manual. NASA TM X-62,282, 1973.

STRUCTURAL OPTIMIZATION APPROACH

- MINIMIZE PLATFORM STRUCTURAL MASS PER UNIT AREA,

$$\left(\frac{M}{A}\right)_{PLATFORM} = \left(\frac{M}{A}\right)_{STRUTS} + \left(\frac{M}{A}\right)_{JOINTS}$$

- WITH RESPECT TO STRUT PROPORTIONS,

  THICKNESSES
  DIAMETERS
  LENGTHS

- SUBJECT TO DESIGN REQUIREMENTS AND CONSTRAINTS.
- OPTIMIZER -- CONMIN COMPUTER PROGRAM.

CHARACTERISTICS OF MINIMUM MASS LEO PLATFORMS UP TO 500 M

(Figure 4)

Optimization results for platforms with diameters, D, up to 500 m are shown in this figure for various constraints. The platforms were required to have a frequency of at least .1 Hz, the struts were required to have a frequency of 10 times the platform design frequency, and the mass of the platform covering was specified to be .1 kg/m^2, which is typical of a low mesh reflector surface. The strut material was graphite-epoxy. Gravity gradient loads were found to be very small. The frequency requirement of the struts sized the struture which resulted in long, small diameter, thin tubes. Minimum mass platforms are characterized by ultra low structural masses (on the order of reflector mesh).

CHARACTERISTICS OF MINIMUM MASS LEO PLATFORMS UP TO 500 M

$f_d = .1\,\text{Hz} \qquad f_s/f_d \geq 10 \qquad m_p = .1\,\text{kg/m}^2$

STRUT FREQUENCY CONSTRAINT DETERMINES SIZE RESULTING IN:

- MINIMUM ALLOWABLE THICKNESSES .5MM (.020 IN.)
- MINIMUM ALLOWABLE DIAMETERS .0127M (.5 IN.)
- LONG LENGTHS 7.38M (24.2 FT.)
- LARGE SLENDERNESS RATIOS AND THUS SMALL AXIAL LOAD CARRYING CAPABILITY

EFFECT OF T/W_0 ON MASS DELIVERED

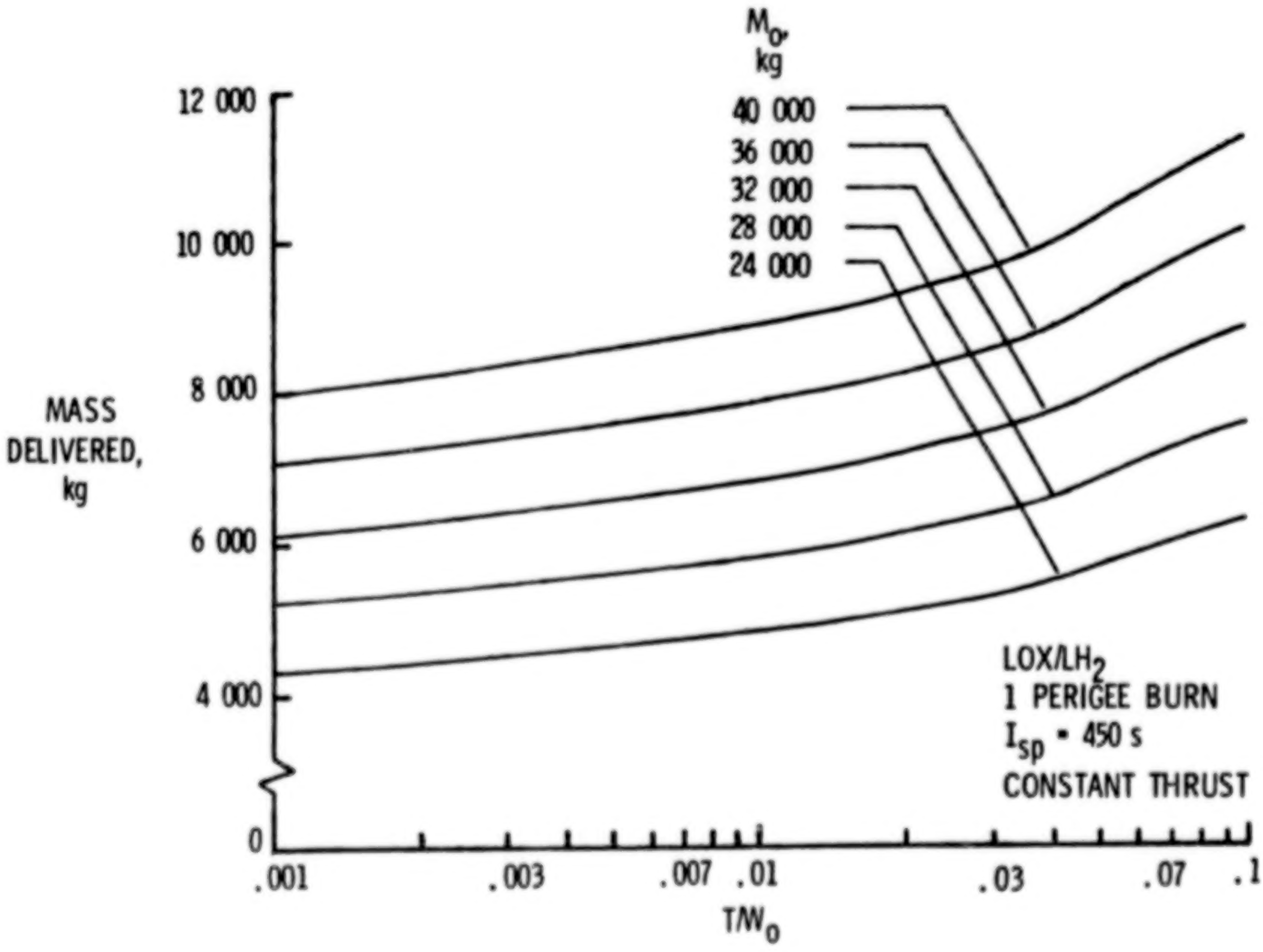
(FIGURE 5)

As a prelude to incorporation of orbital transfer loads, the amount of usable mass that is delivered from LEO to GEO as a function of initial thrust-to-weight ratio is depicted in this figure. In addition the dry mass or mass associated with empty tanks, engines, piping, thrust structure, etc. is also delivered but not shown by these curves. These curves, obtained through the use of the Aerospace Vehicle Interactive Design (AVID) system (ref. 4), are for a liquid oxygen/liquid hydrogen system with constant thrust for one perigee burn. Even though multiple perigee burns increase the amount of usuable payload delivered at the expense of longer trip times, for the initial assessment undertaken here, results for only one perigee burn were developed.

REFERENCE

4. Wilhite, A. W.; and Rehder, J. J.: AVID - A Design System for Technology Studies of Advanced Transportation Concepts. AIAA Paper No. 79-0872, presented at the Conference on Advanced Technology for Future Space System, May 1979.

EFFECT OF T/W_0 ON MASS DELIVERED

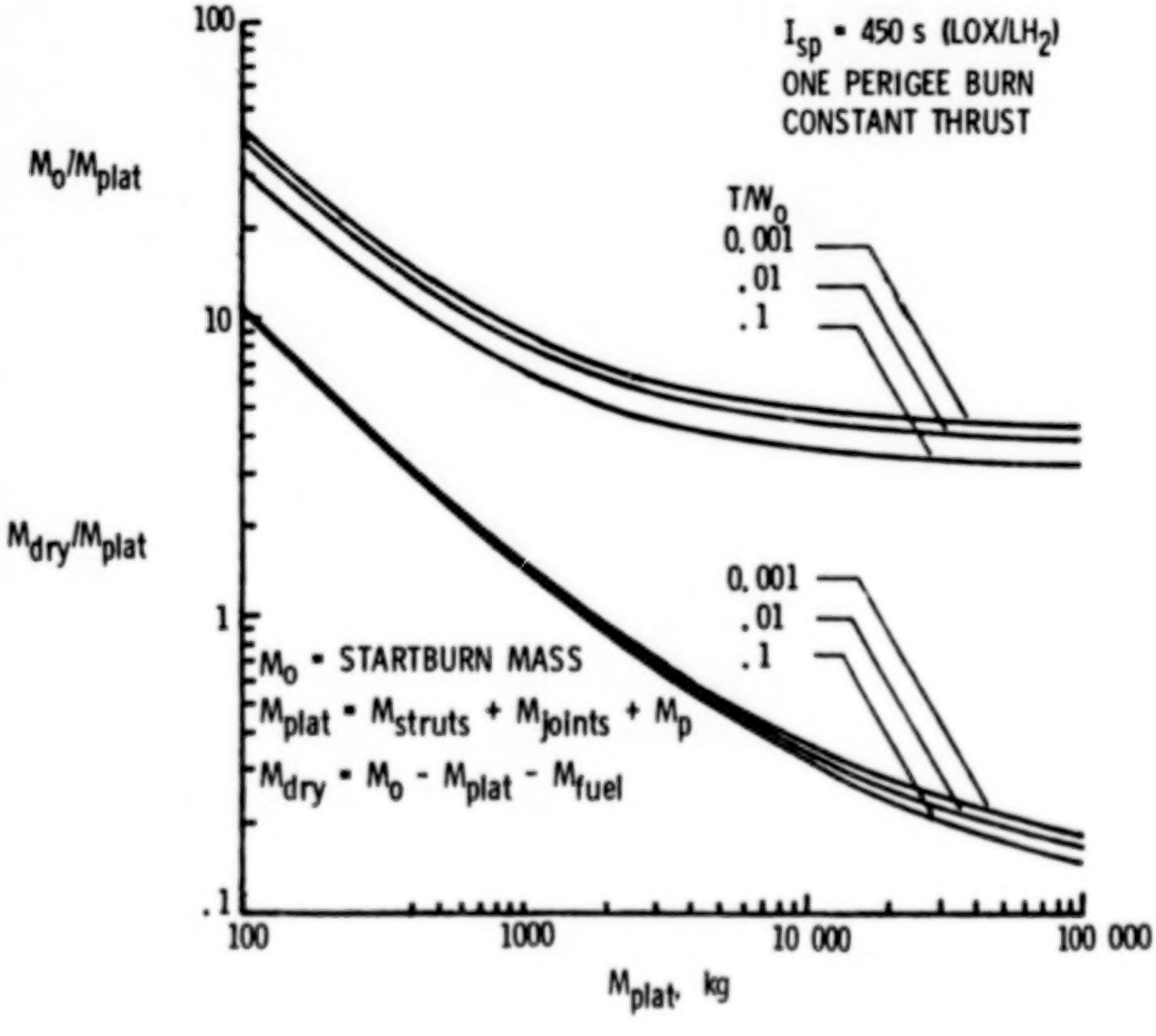


RATIOS OF STARTBURN MASS AND DRY MASS TO PLATFORM MASS AS A FUNCTION OF PLATFORM MASS

(Figure 6)

The information presented in the previous figure can be recast to show the ratio of M_0/M_{plat} as a function of the spacecraft or platform mass. Similarily, the ratio of M_{dry}/M_{plat} as a function of platform mass for selected values of initial thrust-to-weight ratio is shown. This information is incorporated into the sizing procedure. Observe that the mass of the platform contains the distributed mass of the covering, M_p. Also M_0, the starburn mass, is related to the weight, W_0, by g_0 the acceleration of gravity at earth's surface. The motivation for these curves is illustrated in the next figure.

RATIOS OF STARTBURN MASS AND DRY MASS TO PLATFORM MASS AS A FUNCTION OF PLATFORM MASS



ORBITAL TRANSFER THRUST APPLICATION

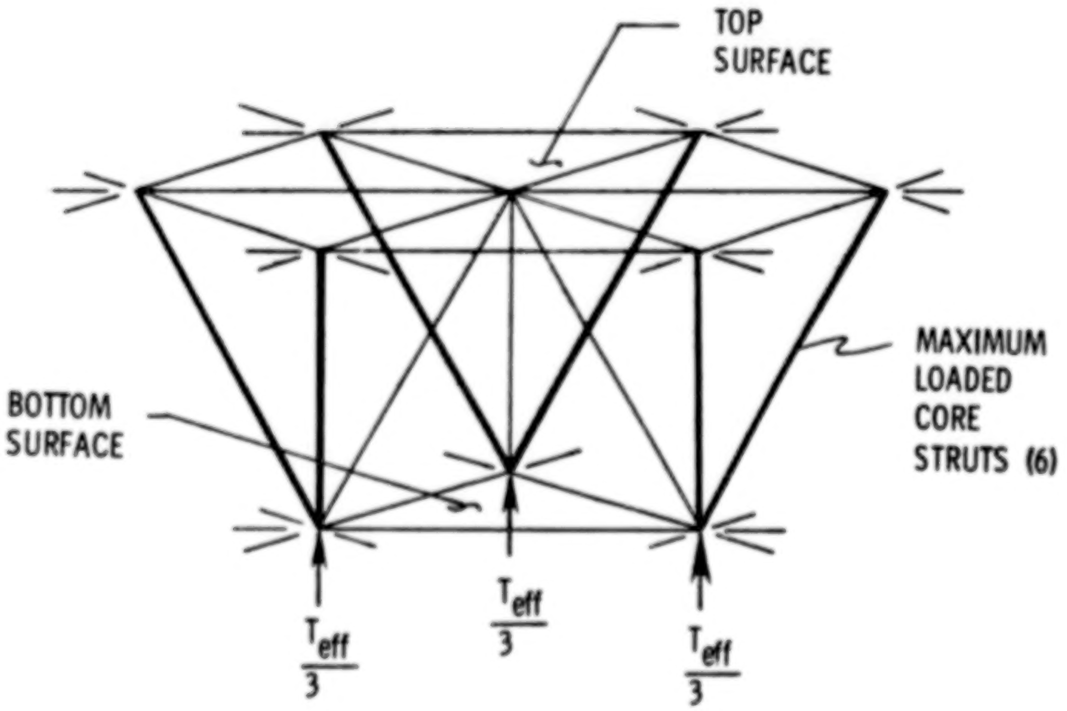
(Figure 7)

A Sketch of the central portion of the tetrahedral truss is depicted in this figure. The thrust load from the engines is introduced at the corners of a centrally located triangle normal to the plane of the back surface. Transient effects of the load were ignored for this initial assessment.

With the struts considered to be pinjointed, the maximum core strut loads occur in six of the nine core struts that connect the bottom triangle to the top surface. The three centermost core struts are essentially unloaded. The remaining six core struts carry the effective thrust load. Effective thrust here means the total thrust minus the dry mass times the final acceleration. The relationship for maximum core load can be manipulated in terms of thrust-to-weight ratio and other mass ratios shown in the previous figure.

For purposes of this sizing study, in which all core struts are identical, all core struts are sized to carry this maximum axial load. The face struts are also sized on the basis of the maximum core struts even though the maximum compressive load in a face strut is less than the maximum core strut load for D/h less than about 25 where h is the depth of the truss.

ORBITAL TRANSFER THRUST APPLICATION



$$T_{eff} = T - m_{dry}\, a_{end}$$

$$Pc_{max} = \frac{1}{6}\,\frac{\ell_c}{\sqrt{\ell_c^2 - \frac{1}{3}\ell_f^2}}\; T_{eff}$$

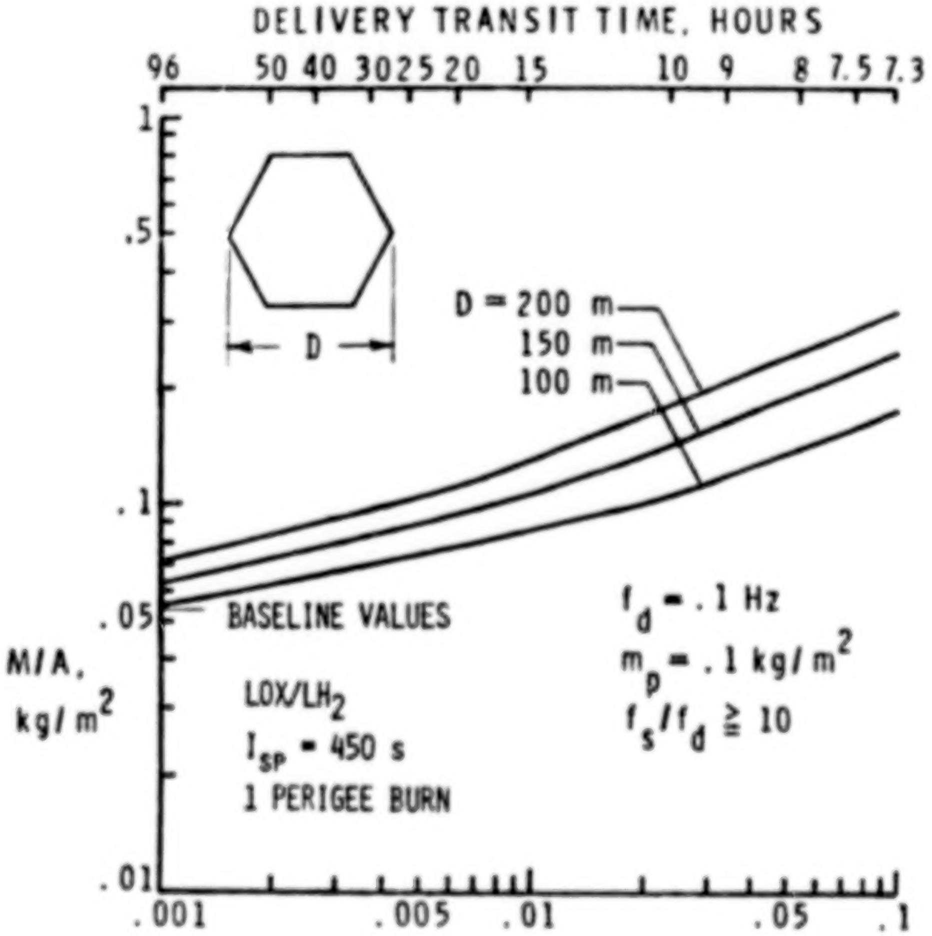
$$Pc_{max} = \frac{1}{6}\,\frac{\ell_c}{\sqrt{\ell_c^2 - \frac{1}{3}\ell_f^2}}\left(\frac{T}{W_o}\right)\left(\frac{M_o}{M_{plat}}\right) M_{plat}\, g_o \left(\frac{1}{1 + M_{dry}/M_{plat}}\right)$$

MASS PER UNIT AREA AS A FUNCTION OF T/W

(Figure 8)

Mass per unit area as a function of initial thrust-to-weight ratio is depicted in this figure for three platform sizes. The propulsion system is assumed to be contained within another Shuttle such that maximum length for the struts is still 18 m. Indicated at the top of the figure is the time it takes for transporting the platform from LEO to GEO. The trusses for GEO application have the same design constraints used previously for LEO platforms. The mass per unit area for the LEO platforms, which is almost identical for the three sizes, is indicated by BASELINE VALUES on the figure.

MASS PER UNIT AREA AS A FUNCTION OF T/W

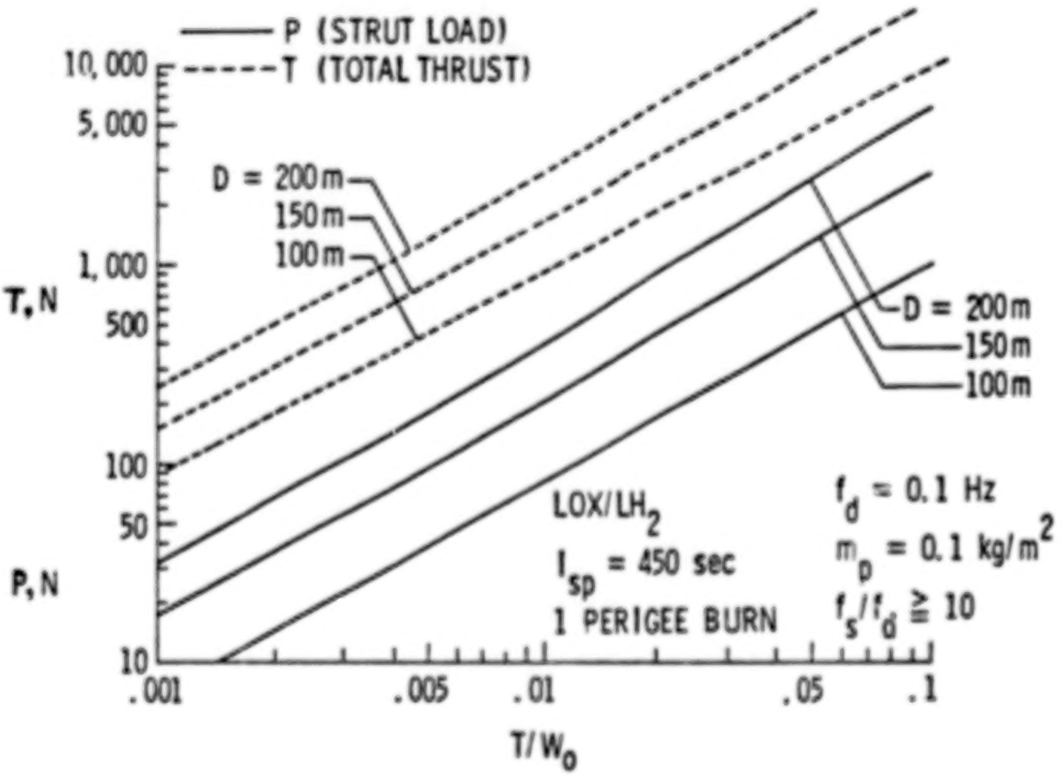


EFFECT OF ORBITAL TRANSFER ON DEPLOYABLE PLATFORM STRUT LOADS

(Figure 9)

The total thrust and maximum core strut load resulting from the chemical propulsion system and design constraints considered are depicted in this figure. The calculations were made without consideration of the availability of a given thrust level engine. The resulting range for thrust is not too different from that being proposed for low thrust chemical engines. Strut are shown to be lightly loaded except for the very highest values of T/W_0.

EFFECT OF ORBITAL TRANSFER ON DEPLOYABLE PLATFORM STRUT LOADS



COMPARISONS OF 100 M LEO AND GEO PLATFORMS

(Figure 10)

This figure compares 100 m diameter platforms sized for LEO and GEO showing the influence of orbital transfer loads. As the thrust-to-weight ratio is increased the minimum mass struts are found to become longer and larger in diameter. They are characterized by minimum gauge thicknesses and exhibit rather large slenderness ratios. In previous figures the parametric results presented did not exhibit an integer number of rings. The reults in this figure are for minimun mass designs constrained to have an integer number of rings.

COMPARISONS OF 100 M LEO AND GEO PLATFORMS

$f_d = .1$ Hz $\quad f_s/f_d \geqq 10 \quad m_p = .1$ kg/m^2

ORBIT	LEO	GEO	GEO	GEO
T/W_0	0.0	0.001	0.01	0.1
T/W_{FINAL}	0.0	0.0036	0.033	0.272
NUMBER OF RINGS	7	7	4	3
$\ell_f \; \ell_c$	7.143 m	7.143 m	12.500 m	16.667 m
$t_f \; t_c$	0.5 mm	0.5 mm	0.5 mm	0.5 mm
$d_f \; d_c$	0.0127 m	0.0127 m	0.0387 m	0.1070 m
f_{plat}	2.77 Hz	2.77 Hz	5.86 Hz	9.29 Hz
f_c	1.16 Hz	1.16 Hz	1.19 Hz	1.86 Hz
NUMBER OF STRUTS	1302	1302	420	234
ℓ/ρ	1591	1591	913	440

COMPARISONS OF 200 M LEO AND GEO PLATFORMS

(Figure 11)

This figure compares 200 m diameter platforms sized for LEO and GEO showing the influence of orbital transfer loads. Many of the same observations about 100 m diameter platmeter platforms hold true. The maximum length for struts is reached at lower values of thrust-to-weight than for 100 m platforms. The frequencies for these larger structures are lower than 100 m platforms and lower values of slenderness ratios are obtained but are still large compared to those of earth based structures.

COMPARISONS OF 200 M LEO AND GEO PLATFORMS

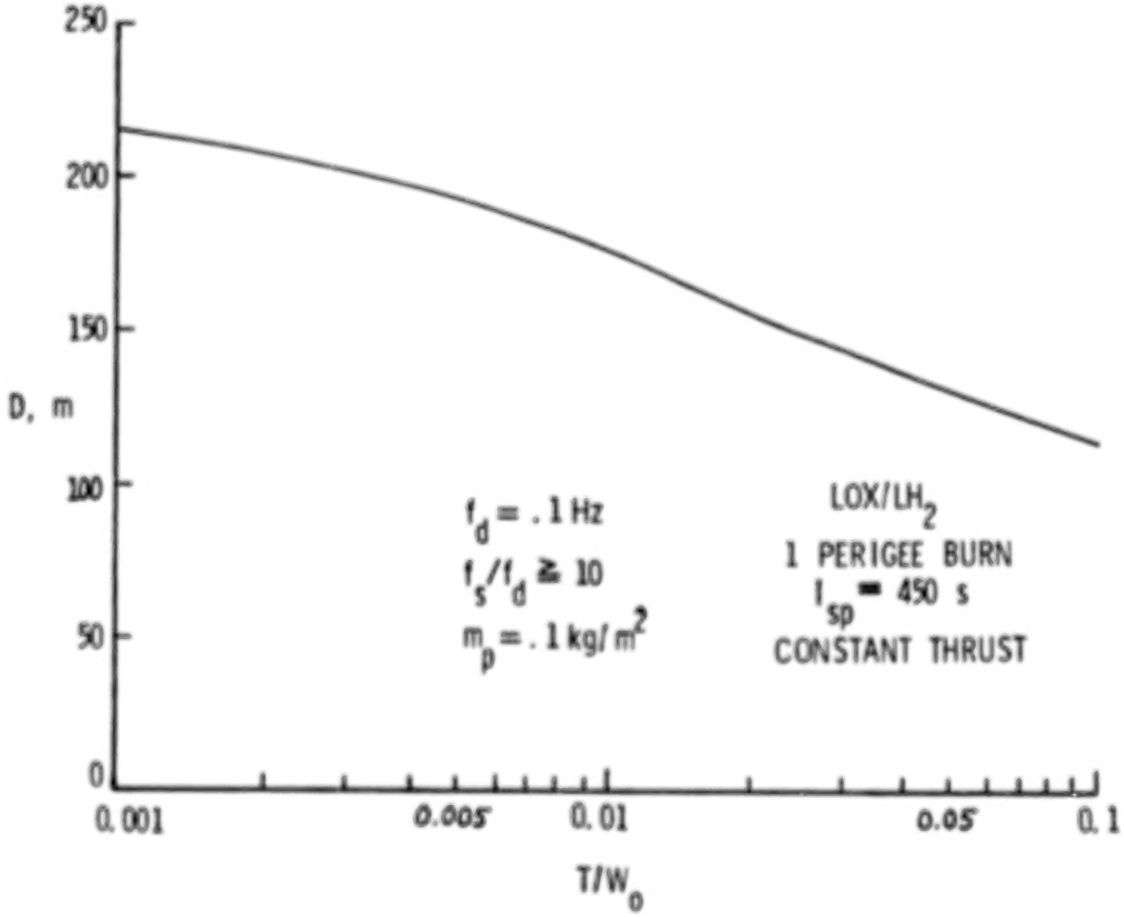
	$f_d = .1$ Hz	$f_s/f_d = 10$		$m_p = .1$ kg/m^2
ORBIT	LEO	GEO	GEO	GEO
T/W_0	0.0	0.001	0.01	0.1
T/W_{FINAL}	0.0	0.0036	0.033	0.272
NUMBER OF RINGS	13	9	6	6
ℓ_f, ℓ_c	7.692 m	11.111 m	16.667 m	16.667 m
t_f, t_c	0.5 mm	0.5 mm	0.5 mm	0.5 mm
d_f, d_c	0.0127 m	0.0263 m 0.0274 m	0.0792 m	0.1953 m
f_{plat}	0.75 Hz	1.22 Hz	2.19 Hz	2.53 Hz
f_c	1.00 Hz	1.02 Hz	1.38 Hz	3.28 Hz
NUMBER OF STRUTS	4524	2160	954	954
ℓ/ρ	1713	1195	595	241

MAXIMUM DIAMETER DEPLOYABLE PLATFORM INCLUDING OTV PACKAGEABLE IN ONE SHUTTLE FLIGHT AS FUNCTION OF T/W_0

(Figure 12)

Up to this point, the sizing procedure generated minimum mass platforms. This figure shows platform size results when the surface area is maximized for the same design constraints used previously. In addition, the mass and volume of the OTV (Orbital Transfer Vehicle) are assumed to package with the structure in one shuttle flight. Since the OTV takes up more than half of the shuttle bay length, only the remaining length is available for packaging the structure. This curve is an upper bound on size because although the distributed non structural or payload mass is considered, the volume associated with its packaging is not.

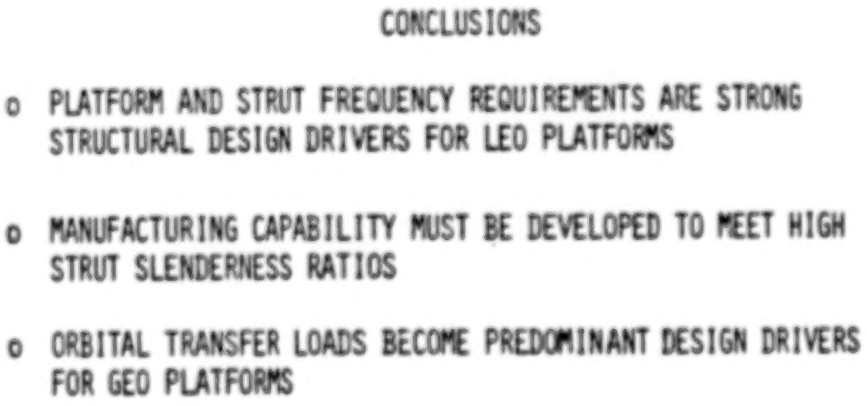
MAXIMUM DIAMETER DEPLOYABLE PLATFORM INCLUDING OTV PACKAGEABLE IN ONE SHUTTLE FLIGHT AS FUNCTION OF T/W_0



CONCLUSIONS

(Figure 13)

For platforms supporting low mass distributed payloads (reflector mesh, etc.), platform and strut frequency requirements (i.e. stiffness) are strong design drivers for LEO applications. The struts are found to be extremely slender, thin-walled, and small diameter. If full advantage is to be taken of these minimum mass designs, a manufacturing capability must be developed for long straight struts. For platforms that are to be transferred from LEO to GEO in a deployed state, the orbital transfer loads become design drivers. However, even for an initial thrust-to-weight ratio equal 0.1, a platform on the order of 100 m in diameter appears packageable with its OTV in one shuttle flight, and larger platforms appear possible at lower thrust-to-weight ratios.



CONCLUSIONS

- PLATFORM AND STRUT FREQUENCY REQUIREMENTS ARE STRONG STRUCTURAL DESIGN DRIVERS FOR LEO PLATFORMS
- MANUFACTURING CAPABILITY MUST BE DEVELOPED TO MEET HIGH STRUT SLENDERNESS RATIOS
- ORBITAL TRANSFER LOADS BECOME PREDOMINANT DESIGN DRIVERS FOR GEO PLATFORMS

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INFLUENCE OF INTERORBIT ACCELERATION ON THE DESIGN OF LARGE SPACE ANTENNAS

John M. Hedgepeth
Astro Research Corporation

INTRODUCTION

Large antennas in space will eventually be needed. Not only will satellite-based communications require antennas of 100 meters or more in diameter but also will remote sensing demand even larger sizes. Some of the predicted needs are characterized in Figure 1, taken from ref. 1. Other studies predict even larger apertures.

Most of the envisioned missions involve orbits that are inaccessible to the Space Shuttle itself. Accordingly, the design of the antenna structure must either countenance automated remote deployment in the operational orbit or must include the loadings due to interorbit boost in the structural requirements of the erected antenna. The purpose of this paper is to investigate, in general, the characteristics of the acceleration-induced loading in structures consisting of triangular lattices and to present some initial quantitative results on the effect on the design mass and stowage volume.

The approach herein is to define the structural design that would be used if no interorbit acceleration were required and then to determine what strengthening would be required to accommodate the loads due to acceleration. The basic zero-acceleration design can be based on the stringent accuracy requirements placed on the antennas.

The missions shown in Figure 1 are seen to involve ratios of diameter to wavelength up to more than 100,000 with the majority centered around a ratio of 1,000. For those missions for which the main beam must contain almost all the radiated energy, the emitted wave front must be accurate to 4 percent of the wavelength. These missions include all the earth-directed antennas in which side-lobe gain must be kept very low. Even in the cases wherein side-lobe gain is of primary importance, the rms errors in the wave front are held to less than 12 percent of the wavelength. These missions include outward-pointed antennas for which the side-lobe gain can be relatively large.

In a reflector antenna, the wave-front error is very nearly twice the component of structural distortion normal to the reflector surface. Thus, the surface error of a reflector antenna must be held to one-fiftieth of a wavelength for the low-side-lobe missions and one-sixteenth of a wavelength for the high-gain missions.

Combining the foregoing relationships with the data in Figure 1 yields the requirement on structural surface accuracy. Submillimeter radio astronomy, for example, requires an accuracy of one part per million of the diameter. Those earthward-pointed missions which have a diameter wavelength ratio of around 1,000 require a surface accuracy of 20 parts per million. At the other end, low-frequency radio astronomy allows the surface error to be as much as one-thousandth of the diameter.

STRUCTURAL CONFIGURATIONS

The type of spacecraft under consideration is shown in the center of Figure 2. It consists of a reflector and a radiofrequency feed mounted at a distance by some sort of structure. Of course, the feed position and orientation with respect to the reflector is important, but in this paper attention is confined to the reflector portion only.

Four reflector configurations are shown in Figure 2 and in more detail in Figures 3 through 6. These four are selected to encompass the types that utilize a knitted mesh material for the actual reflector surface. Such material packages very well, is lightweight (~50 g/m^2), is compliant, and only needs to be positioned properly to be an excellent reflector.

The tetrahedral truss has been discussed by many authors. Differences exist in scale and in the manner in which the structure and the mesh interface. In the form treated herein, the interface with the mesh is only at the triangular lattice nodes. Separate tendons under high tension are laced through the mesh along lines parallel to the surface truss elements and attached at the nodes. The structural members therefore must carry only axial compression and tension and can thus be slender for lightly loaded situations. Properly located joints allow stowage and deployment of the otherwise uncompliant structure. From an overall standpoint, the tetrahedral truss structure can be thought of as a thick shell, the surface of which is defined by the lattice nodes. For the equilateral triangular geometry, the shell is isotropic, an advantage that does not obtain for some of the other truss geometries proposed.

The geodesic dome can be viewed as the limiting case of a tetrahedral truss as the thickness H is reduced to zero. The geodesic dome behaves in the large as a membrane. It is simpler than the truss since only one surface of lattice elements is required. On the other hand, the membrane-like surface is very flexible unless the edge is supported by a stiff ring. Packaging and deploying the ring may present more difficulties than those presented by the more nearly uniform tetrahedral truss. The interface with the mesh is again assumed to be at the lattice nodes and the structural members carry axial tension and compression only.

The radial-rib configuration has as its structure a large number of radially oriented curved beams that are cantilevered from the central hub. The interface with the reflecting mesh is continuous along the chords of the beams. Thus the mesh is in gores rather than facets as is the case for the other configurations. The beams are stowed by wrapping them around the central hub with the necessary compliance supplied in a number of ways. The ATS-6 antenna is a salient example of this configuration.

The pretensioned truss is the author's version of the variously named "Maypole," "Hoop-and-Column," "Wire-Wheel," and "Spoked-Wheel" concepts. The basic structural element is the bicycle-wheel structure made up of the central column (hub) and the compression rim tied together by stays. The rim is articulated, allowing stowage.

The central column is an Astromast. The rest of the structure is "soft" in the sense that its elements need to carry tension only. Thus a variety of packaging techniques can be used without requiring complex joints. On the other hand, the deployed structure is "stiff." The tension-carrying elements are pretensioned sufficiently to allow incremental compression loading in orbit while still retaining positive tension. The front and back stays, for example, thus maintain their full axial stiffness.

The reflector surface is formed by structural tension-stiffened radial beams. The tension in the curved chords automatically pretension the interchord members. The chord pretension is reacted by the compression rim. A compression spreader is needed at the outer end. The pretensioned beam is cantilevered at the central hub and also supported at the tip by the rim. Circumferential tension members provide the remainder of the structure. They and the upper chords of the beams are laced through the mesh to provide the necessary shaping to the reflector surface in quadrilateral facets.

MESH SADDLING

Since the mesh has no bending stiffness, it behaves like a membrane; it can carry no compression. Furthermore, the tension must be sensible and reasonably uniform and isotropic in order to assure good electrical conductivity (and, hence, rf reflectivity) of the mesh. Values of around 2.5 N/m are used, for example, in the Harris studies in ref. 2.

A biaxially tensioned membrane with no lateral loading must have zero Gaussian curvature. Thus if the curvature in one direction is positive, the curvature in the other direction must be negative. Desired reflector surfaces are approximately spherical with a radius of curvature of twice the focal length F. Unfortunately, mesh surfaces want to look like saddles.

For a faceted reflector configuration, the best approximation to a dish is to make the facets flat, with the corners located so as to cancel the average deviation between the flat and the desired curved surface. The rms deviation is kept small enough by limiting the size of the facets.

At the intersection between adjacent facets, the tension in the mesh changes direction. This produces a slight bow of the supporting tendon laced through the mesh as illustrated in Figure 7. The deviation between the saddled mesh surface and the desired spherical surface is

$$w = \frac{1}{12F}\left\{\frac{\ell^2}{4} - 3x^2 - 3y^2 + \frac{2N}{T}\left[\frac{\sqrt{3}}{90}\,\ell^3 + x(x^2 - 3y^2)\right]\right\}$$

where x and y are Cartesian coordinates with origin at the center of the triangle and the negative x axis passing through a vertex. The mesh tension per unit length is N and the tendon tension is T. The corresponding rms deviation divided by the antenna diameter is

$$\frac{w_{rms}}{D} = 0.01614\,\frac{(\ell/D)^2}{F/D}\left(1 + 0.33\,\frac{N\ell}{T}\right)$$

In order to allow the largest facet size, the tendon tension must be large, say greater than 10 $N\ell$. Then the facet size for an allowable value of rms deviation is

$$\frac{\ell}{D} = 7.87\sqrt{\frac{F}{D}\left(\frac{w_{rms}}{D}\right)_{\text{Allow.}}}$$

for the triangular facet.

If the facets are rectangular, the same process yields

$$\frac{w_{rms}}{D} = 0.0186\,\frac{(\ell/D)^2}{F/D}\sqrt{1 + \left(\frac{b}{\ell}\right)^4}\left[1 + C\left(\frac{b}{\ell}\right)\frac{N\ell}{T}\right]$$

where b is the smaller rectangle dimension, and C varies from about 0.2 to about 0.6 as b/ℓ varies from 1 to 2. Again, in order to maximize facet size, set $b < 0.5$ and $T > 10N$. Then

$$\frac{\ell}{D} = 7.33\sqrt{\frac{F}{D}\left(\frac{w_{rms}}{D}\right)_{\text{Allow.}}}$$

for the rectangular facet.

For the radial-rib configuration, the mesh is in gores. The curvature in the radial direction is enforced to be that of the rib. The saddling produces a negative curvature in the circumferential direction equal to N_1/N_2 times the radial curvature, where N_1 and N_2 are the membrane tensions in the radial and circumferential directions, respectively (see Figure 7). The resulting rms deviation is

$$\frac{w_{rms}}{D} = 0.01076\,\frac{(\ell/D)^2}{F/D}\left(1 + \frac{N_1}{N_2}\right)$$

where ℓ is the gore width at the rim. For isotropic mesh tension, $N_1 = N_2$ and

$$\frac{\ell}{D} = 6.82\sqrt{\frac{F}{D}\left(\frac{w_{rms}}{D}\right)_{\text{Allow.}}}$$

for the gore configuration.

The facet and gore sizes are shown in Figure 8. These curves can be used to determine the required degree of refinement of the structural geometry.

EFFECT OF FABRICATION IMPERFECTIONS

Designing the geometry correctly is only the first step. The departure of the as-fabricated structure from the design must also fall within acceptable limits. Presumably, the effects of systematic fabrication imperfections can be removed by a combination of tooling and testing. There still remains the surface error due to random imperfections.

This subject is treated in detail in ref. 3. The results are characterized in Figure 9. In this figure, the achievable ratio of diameter to wavelength is shown as a function of the standard deviation of the unit length error σ_ε of the members composing the structure for various structural configurations. Note that the radial-rib design is not included because of its much lower potential capability.

The quantity σ_ε is at the control of the designer, although with a considerable cost impact. In general, a value of σ_ε of 10^{-3} is representative of ordinary careful practice, of 10^{-4} is characteristic of a high-quality machine ship, of 10^{-5} is achievable with well designed and operated hard tooling, and of 10^{-6} is very difficult and costly.

The difficulty in achieving very small values of σ_ε can be visualized by considering to what stress levels they correspond. For example, in steel, magnesium, titanium or aluminum, the stress level corresponding to a strain of 10^{-6} is induced in only 2.5 meters of material vertically suspended in a 1-g field.

In preparing Figure 9, the criterion was established that the surface distortion shall be limited by one-half of the allowable $\lambda/50$ that is the requirement for most of the missions described in Figure 1. This is done in order to allow the various sources of error (which are, in general, additive on a mean-square basis) to coexist and still be able to meet the $\lambda/50$ requirement.

A particular ratio of focal length to diameter of two is chosen for the comparison. Most antennas with electronically steerable beams will require such a high F/D.

As can be seen in Figure 9, the tetrahedral truss is by far the most attractive configuration for attainment of large apertures with acceptable error due to fabrication imperfections. A value of D/λ of nearly 10,000 is possible for a fabrication tolerance parameter of 10^{-5}. Reference to Figure 1 shows that this ratio would encompass all the missions except those involving submillimeter and IR astronomy. And if the relaxed λ/16 criteria were used, a value of D/λ = 30,000 would be feasible. Thus even submillimeter astronomy is possible from this standpoint.

The pretensioned truss is probably more readily packaged than the tetrahedral truss. It shows good accuracy for most of the missions.

Even the geodesic dome and a deep-rib design present usable accuracy for the smaller-aperture communication-satellite missions.

ENVIRONMENTAL STRAINS

The antenna must remain accurate in the presence of environmental effects after it is established in space. It is assumed that materials will be available with the necessary dimensional stability in the vacuum, UV, and particulate radiation environment that exists in orbit. Furthermore, it is assumed that redundant design will be used to resist the deleterious effects of the uncertainty in such strains can be kept to acceptable limits by proper design. (Indeed, this latter requirement is probably the overriding design criterion.) But there remains the ubiquitous effects of thermal strains.

The influence of thermal strains on surface accuracy is complex and dependent to a great extent on detailed design. Some overall preliminary considerations are considered in ref. 3. Attention is restricted to the tetrahedral-truss structure inasmuch as it exhibits the most potential for accurate reflectors. The results are summarized in Figure 10.

Here the ratio of diameter to wavelength is shown as a function of the maximum thermal strain parameter $\alpha_T T_{max}$, where α_T is the thermal expansion coefficient and T_{max} is the maximum radiation equilibrium temperature for a general member.

When the sun shines on a triangular grid of elements, some of them are hotter than the others because their axes are more nearly normal to the solar radiation. The differing temperatures cause differing strains in the members of differing orientation. The strains can be expressed in terms of equivalent biaxial normal and completely defined by the average strain ε_{ave} and the maximum shear strain γ_{max}. Results for the effects of average and shear strain are shown in Figure 10.

Another source of thermal gradient is the temperature difference between the two faces of the tetrahedral truss due to shading on one face by the other - and by the intersurface members. The amount of shading depends, of course, on the slenderness of the truss members. (Note that shading due to the mesh is assumed to apply uniformly to both surfaces.) The analysis is linearized with respect to d/ℓ and is therefore only accurate for low d/ℓ. It considers only shading due to the surface members. The shading due to the intersurface members is included approximately by the factor k in the expression for the strain differential.

The maximum shading effect is obtained when the sun strikes the surface perpendicular to a set of members. Total blocking is achieved for glancing illumination. Of course, this situation is unrealistic for the curved dishes under consideration. For this reason, the curves are cut off at $\theta = 80°$.

The temperature differences between surfaces could be a severe limiter on the antenna sizes for the tetrahedral truss, the effects being much more severe than either overall temperature-strain effects or shear-strain effects. For a nominal worst case of $T_{max} = 295$ K and α_T of 0.5×10^{-6}/K (readily achieved for graphite/epoxy), the limiting value of D/ℓ is 1,000. In order to achieve the D/λ of 10,000, of which the tetrahedral truss is otherwise capable, an order of magnitude improvement would be required. This could be accomplished through a combination of deepening the truss, making the members more slender (perhaps not feasible if loading is already high), reducing the absorptivity-emissivity ratio, and finally, assuming a more stable material. Much remains to be done in this area.

LOADS DUE TO INTERORBIT ACCELERATION

Consider a tetrahedral truss dish of circular plan form which is accelerated by a thrust at its center of gravity. The thrust is applied perpendicular to the dish. For most antennas, the dish is shallow enough and the facet size is small enough that the tetrahedral truss will behave like a flat plate insofar as overall deformation and loading are concerned. The radial and circumferential bending moments so produced must be absorbed by radial and circumferential stress restraints in the upper and lower truss surfaces as follows:

$$N_r = \mp \frac{1}{16} m_p D\ddot{z}\left(1 + \frac{m_s}{m_p}\right) \frac{D}{H} \left[\frac{3 + \nu}{4}\left(1 - \frac{4r^2}{D^2}\right) + (1 + \nu)\ln\frac{2r}{D}\right]$$

$$N_\phi = \mp \frac{1}{16} m_p D\ddot{z}\left(1 + \frac{m_s}{m_p}\right) \frac{D}{H} \left[\frac{5\nu - 1}{4} - \frac{1 + 3\nu}{4}\,\frac{4r^2}{D^2} + (1 + \nu)\ln\frac{2r}{D}\right]$$

where D and H are the diameter and depth of the dish, r is the radial coordinate, $\ddot{z}$ is the acceleration, m_p is the mass per unit area of the nonstructural payload (the mesh for the antennas under consideration herein) and m_s is the mass per unit area of the structure. Note that Poisson's ratio ν is equal to 1/3 for equilateral triangular lattices.

For the upper surface, an additional uniform isotropic compression induced by the mesh itself must be added to the foregoing acceleration-induced loads.

The shear load resultant, which must be carried by the intersurface struts, is

$$Q_r = \frac{m_p D\bar{z}}{4}\left(1 + \frac{m_s}{m_p}\right)\left(\frac{2r}{D} - \frac{D}{2r}\right)$$

For the geodesic dome, assume that the thrust is applied at the feed position and is carried into the reflector structure through the rim. Then the radial and circumferential stress resultants in the dome surface are:

$$N_r = -m_p D\bar{z}\left(1 + \frac{m_s}{m_p}\right)\frac{F}{D}\left(1 + \frac{r^2}{16F^2} + \frac{r^4}{128F^4}\right) - N$$

$$N_\phi = -m_p D\bar{z}\left(1 + \frac{m_s}{m_p}\right)\frac{F}{D}\left(1 - \frac{5}{16}\frac{r^2}{F^2} - \frac{3}{128}\frac{r^4}{F^4}\right) - N$$

where F is the focal length. Note that this expression includes the additional loading N induced by the mesh.

For the triangular lattices under consideration herein, these surface loadings can be converted into design loads on the individual structure members. The strut loadings are dependent on the orientation of the triangular lattice with respect to the principal directions of loading. Thus, the strut compression is

$$P = -\frac{\sqrt{3}}{3}\ell\left[\frac{N_r + N_\phi}{2} + (N_r - N_\phi)\cos 2\alpha\right]$$

where α is the angle between the member and the direction of N_r. The maximum compressive load is given by $\alpha = 0$ or $\pi/2$ and is

$$P = \frac{\sqrt{3}}{3}\ell\left[|N_r - N_\phi| - \frac{N_r + N_\phi}{2}\right]$$

For the geodesic dome, the resulting worst-orientation strut loads are as shown in Figure 11. For the tetrahedral truss, the worst-orientation strut loads due to acceleration are shown in Figure 12. Note that in the case of the geodesic dome, the loading is dependent on the focal-length-diameter ratio both explicity in the equation and in the parameter C.

The compressive loads in the intersurface struts are also dependent on their orientation. The worst-orientation load is

$$P = \frac{\ell m_p D \bar{z}}{4}\left(1 + \frac{m_s}{m_p}\right)\sqrt{1 + \frac{1}{3}\,\frac{\ell^2}{H^2}}\left(\frac{2r}{D} - \frac{D}{2r}\right)$$

STRUT SIZING

Each strut is assumed to be a thin-walled hollow tube with a wall thickness t and is designed to carry the compression load P as an Euler column with a factor of safety of F.S. The resulting diameter-to-length ratio of that strut is

$$\frac{d}{\ell} = \left(\frac{8F.S.}{\pi^3}\,\frac{P}{t\ell E}\right)^{1/3}$$

where E is Young's modulus. The mass per unit area of a single surface of these struts is

$$\frac{\text{Mass}}{\text{Area}} = 4\sqrt{3}\,k\rho\left(F.\,S.\,\frac{Pt^2}{E\ell}\right)^{1/3}$$

where ρ is the density of the strut material and k is a factor which is introduced to include the mass of the fittings.

Conceptually, it would be possible to design each separate strut with a proper diameter to carry the loading at its particular location and orientation. From a practical point of view, the fabrication problems involved in having many different sizes of members are undesirable. Therefore, in the results herein, the assumption is made that all members are the same for the geodesic dome, for example. Thus the struts are designed to carry the maximum compression loads at the rim.

For the tetrahedral truss, the upper surface struts have the maximum loading at about the 80-percent radial station. It is assumed that all upper surface members are sized to carry this load. On the lower surface, the same size of struts are used as those of the upper surface unless the loading gets higher than their design load. As the center is approached, therefore, larger struts will be required. They are assumed to be all sized in accordance with the loading at the 5-percent radial station. The structure inboard of that station is considered to be thrust structure which is specially designed and is part of the propulsion system. Finally, the intersurface struts are assumed to have the same cross section as the lower surface struts.

Of course, the foregoing procedure of designing for compression is based on the assumption that tension strut loads are easily carried so that they have no effect on the design. This is indeed the case for such lightly loaded structures.

DESIGN MASS AND STOWAGE VOLUME

For the geodesic dome and the tetrahedral truss structure, the structural mass per unit area for the zero-acceleration case is

$$(m_s)_0 = 4\sqrt{3}\rho\left(\frac{\sqrt{3}}{3}\frac{Nt^2}{E}\text{F.S.}\right)^{1/3} \quad \text{(Geodesic Dome)}$$

$$= 4\sqrt{3}\rho\left(\frac{\sqrt{3}}{3}\frac{Nt^2}{E}\text{F.S.}\right)^{1/3}\left(2+\sqrt{\frac{1}{3}+\frac{H^2}{\ell^2}}\right) \text{(Tetrahedral Truss)}$$

The stowage length for "standard" packaging in which each strut is hinged in the middle is nominally $\lambda/2$ for the geodesic dome and $\ell + \sqrt{\ell^2/3 + H^2}$ for the tetrahedral truss.

The ratio of stowage diameter to deployed diameter is $3d/\ell$ for the "standard" packaging. In order to avoid problems from nonuniformities, it should be assumed that the joints are constructed with the outer diameter of the largest strut even when used with smaller struts.

INFLUENCE OF INTERORBIT ACCELERATION

Results for the increase in average mass per unit area and stowage diameter ratios are shown in Figure 13 for the geodesic dome and Figure 14 for the tetrahedral truss. For these examples, the required reflector mesh tension is assumed to 2.5 N/m (the geometric mean of 1.75×3.5 N/m, see ref. 2) and a support-tendon multiplier of 10 is used. Thus, N = 25 N/m. The tube wall thickness is selected to be 0.35 mm, the factor of safety to be 2, and the fitting factor to be 1.5. The material is assumed to be graphite/epoxy with a modulus of 110×10^9 N/m^2 and a density of 1520 kg/m^3, with a resulting structural unit mass as given in the figures. In the case of the tetrahedral truss, the depth and the surface-strut length are assumed to be equal and of the value shown in Figure 14, which is appropriate to a surface-accuracy budget of 10^{-5}.

In Figures 13 and 14, the unit structural mass and the diametral stowage ratio are given as a function of the interorbit acceleration for several diameters. The geodesic dome is very tolerant of acceleration, probably because the rim is used to distribute the load. Note that the results are for the dome portion only and do not include the mass or stowage volume for the rim.

The tetrahedral truss exhibits great sensitivity. Even the "small" 100-m-diameter reflector suffers a 50-percent increase in structural mass and a 100-percent increase in stowage diameter at an acceleration of 1 m/sec^2. Note, however, that the packaged 100-m dish still weighs less than 2300 kg and has a diameter of 3.5 meters and a length of 7.6 m.

The results, of course, are only illustrative. No attempt has been made to seek high structural efficiency. A considerable reduction in the influence of acceleration could be attained simply by tailoring the strut selection to its particular orientation, even if only two sizes were used. Even more reduction could be achieved by using more than two sizes.

Similarly the simplest of basic strut designs has been used. The structure is heavy. For the tetrahedral truss, it is more than three times the weight of the payload (the mesh). Obvious potential exists for weight reduction.

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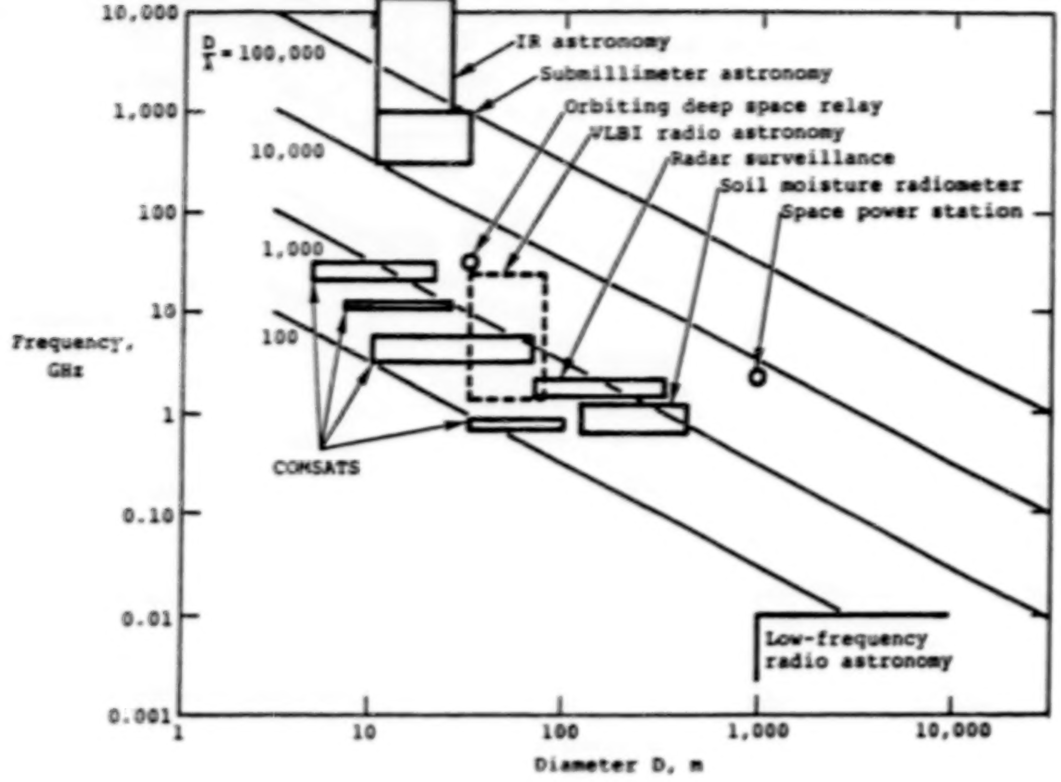
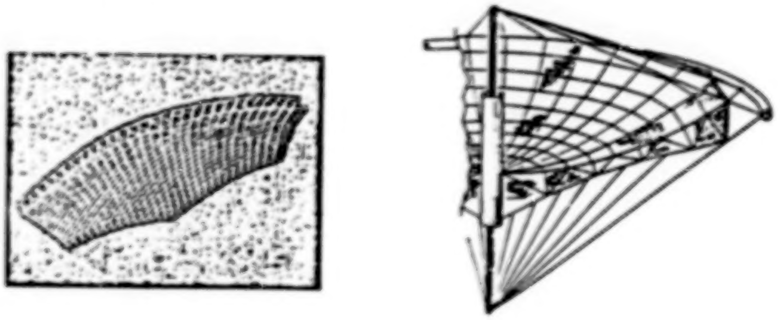
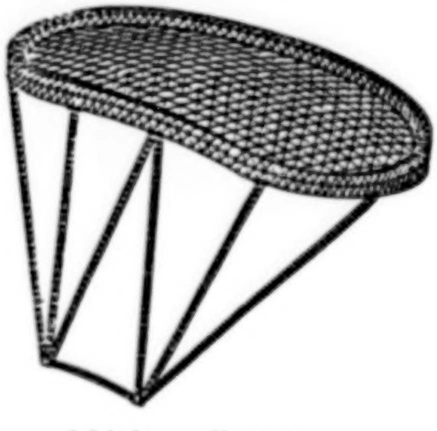


Figure 1. Large space antenna requirements.

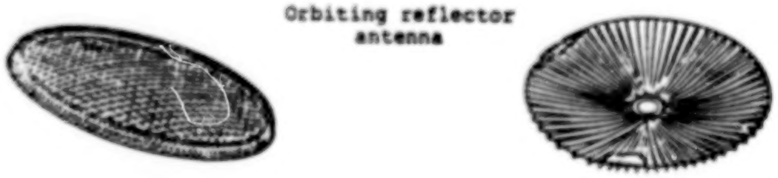


Tetrahedral truss

Pretensioned truss



Orbiting reflector antenna



Geodesic dome

Radial rib

Figure 2. Antenna structural configurations for mesh-type reflectors.

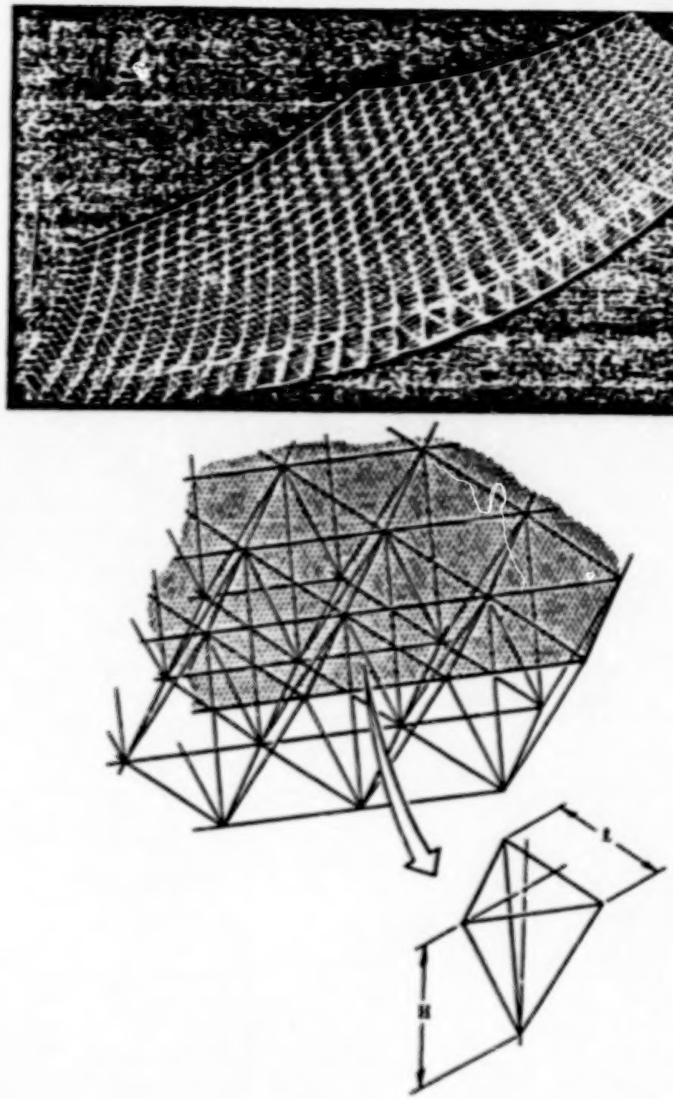


Figure 3. Tetrahedral-truss configuration.

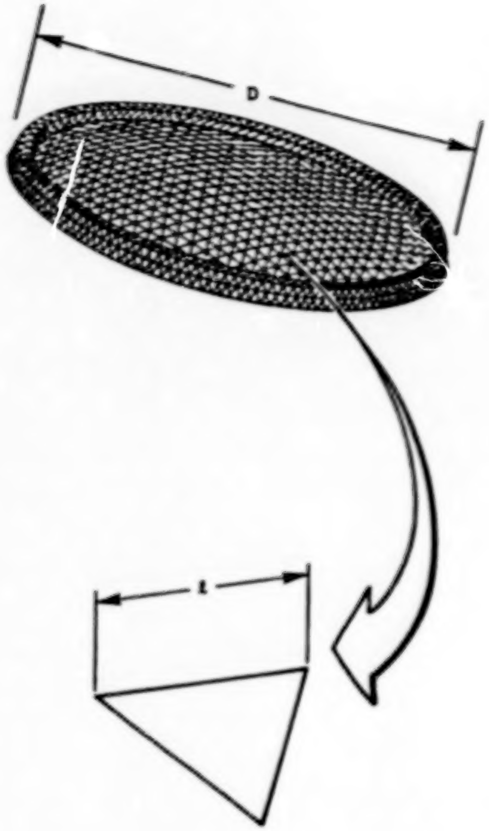


Figure 4. Geodesic-dome configuration.

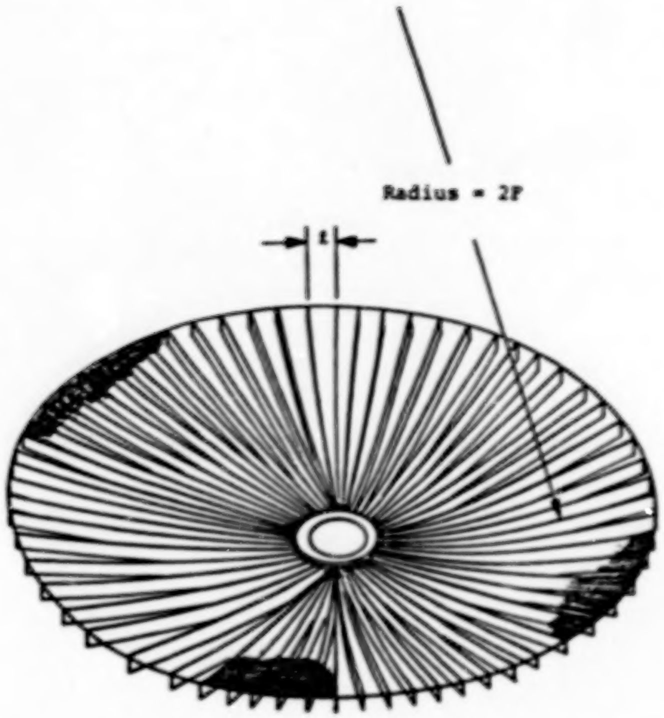


Figure 5. Radial-rib configuration.

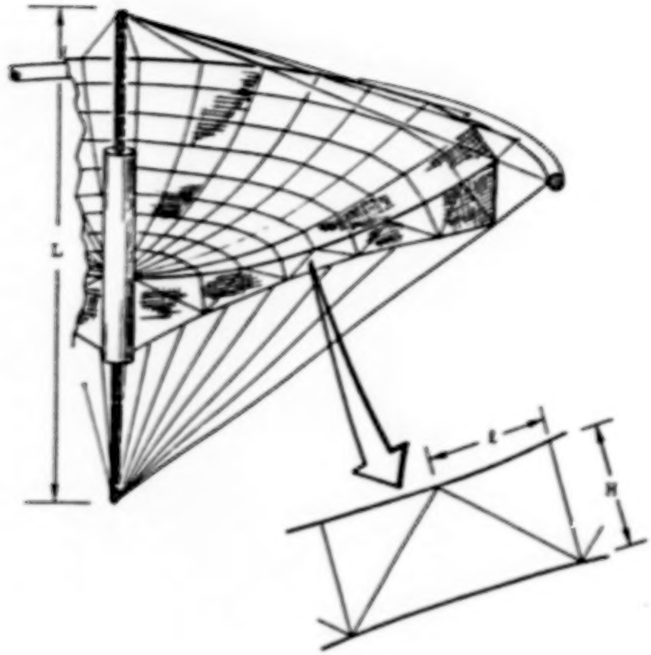
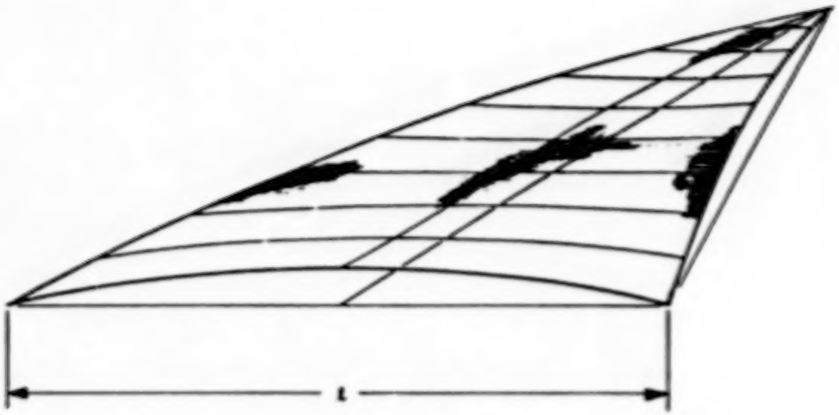


Figure 6. Pretensioned-truss configuration.



Triangular facet

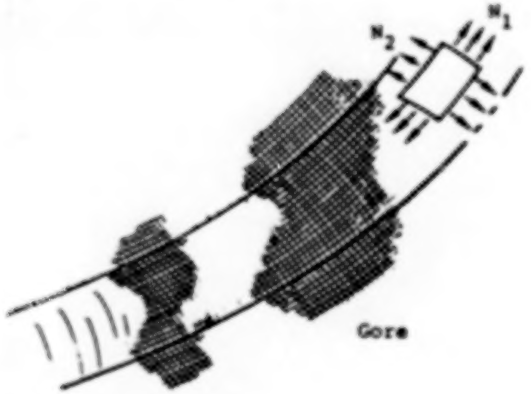


Figure 7. Mesh saddling.

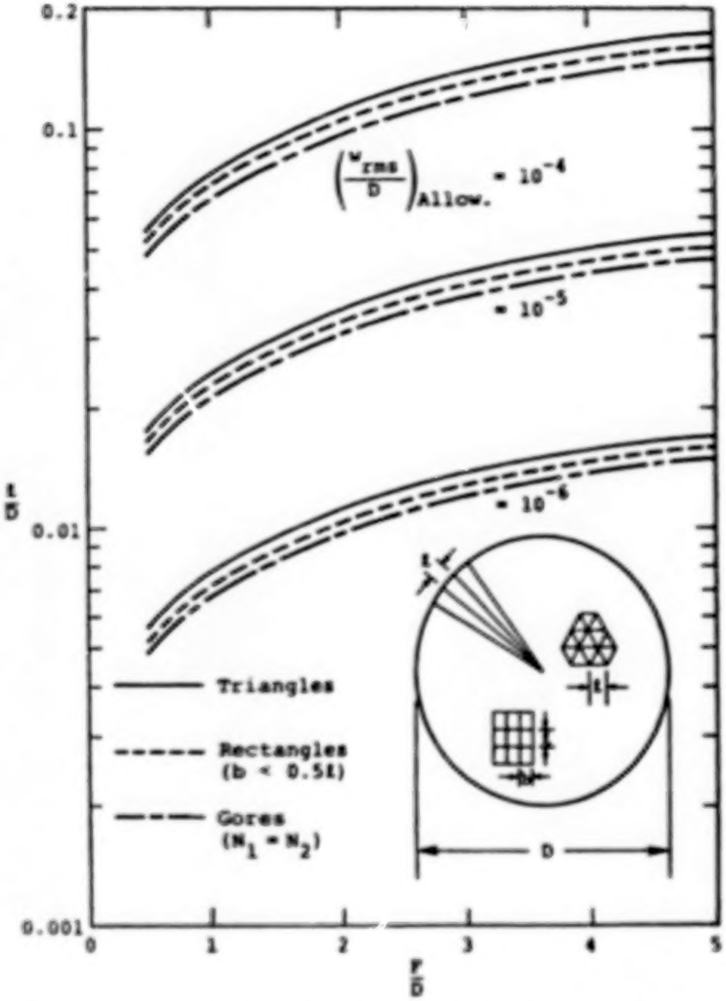


Figure 8. Influence of focal-length ratio and allowable surface error on size of mesh panels.

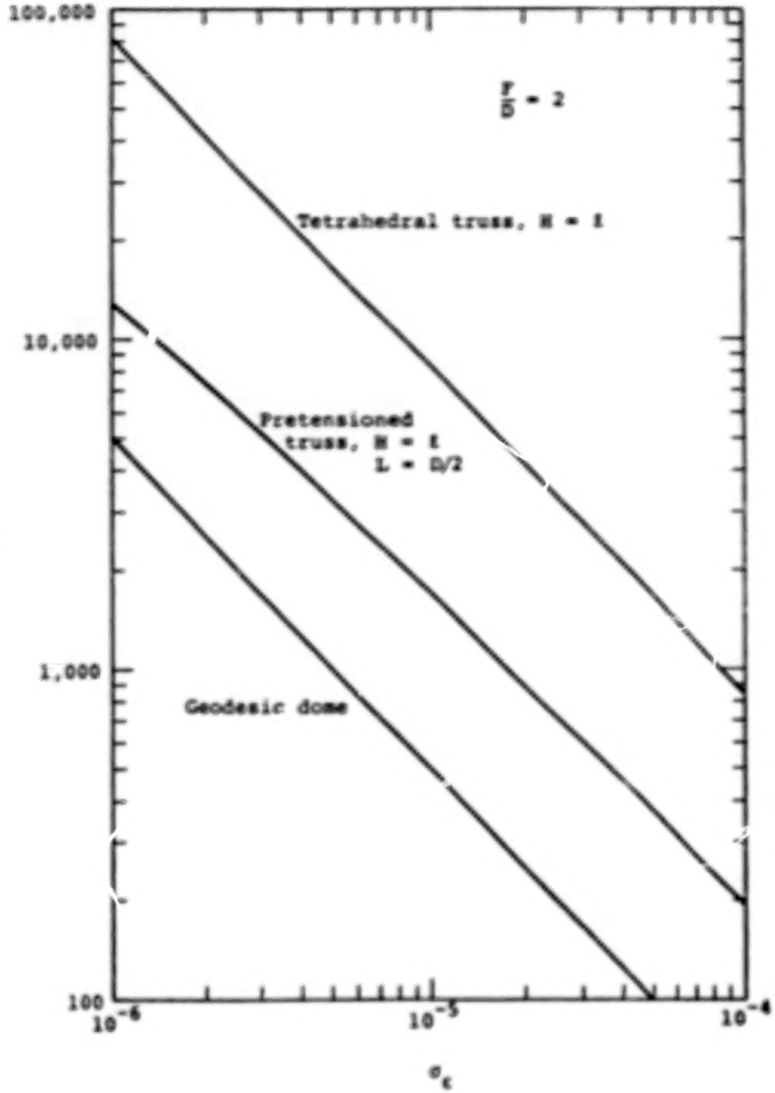


Figure 9. Potential antenna sizes for $w_{rms} = \lambda/100$ as limited by fabrication imperfections.

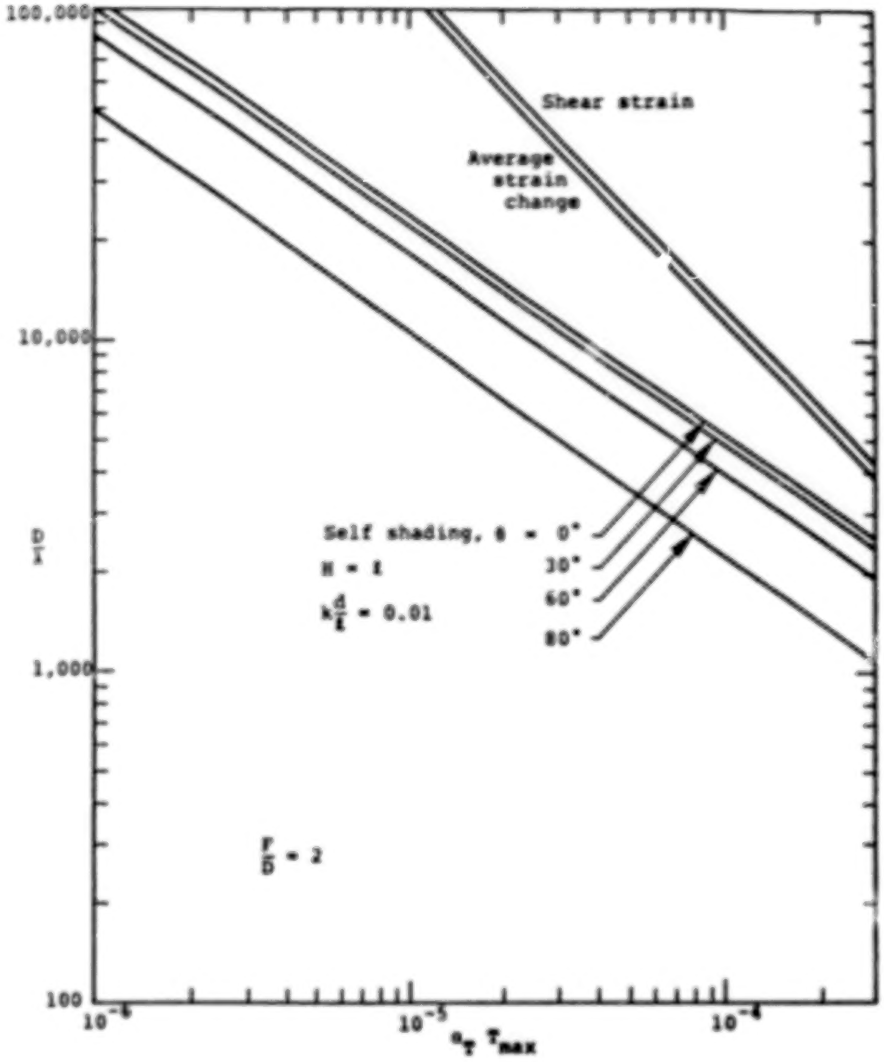


Figure 10. Potential tetrahedral-truss antenna sizes for $w_{rms} = \lambda/100$ as limited by thermal strains.

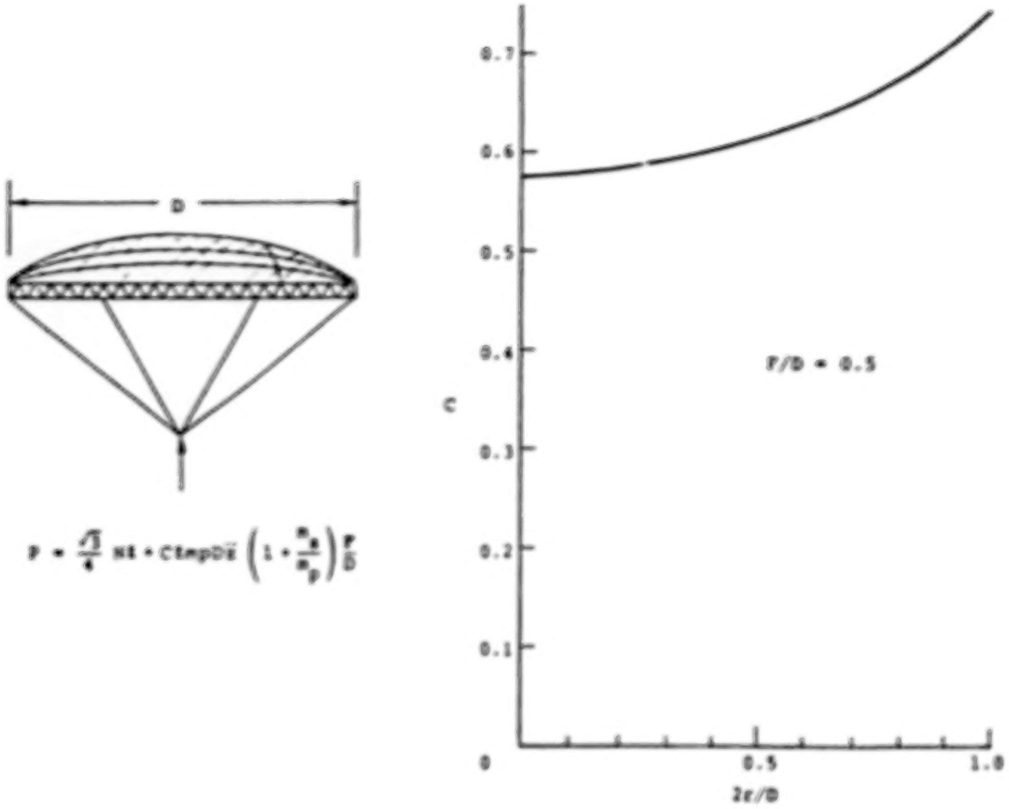


Figure 11. Strut loads in geodesic dome.

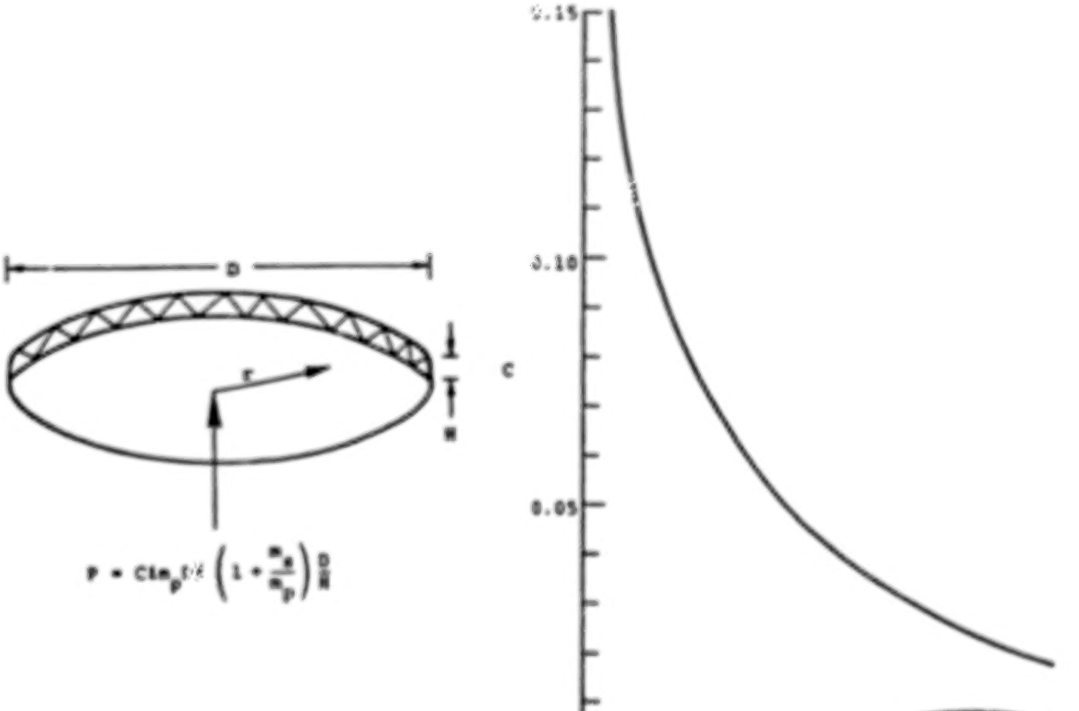


Figure 12. Strut loads due to acceleration - tetrahedral truss.

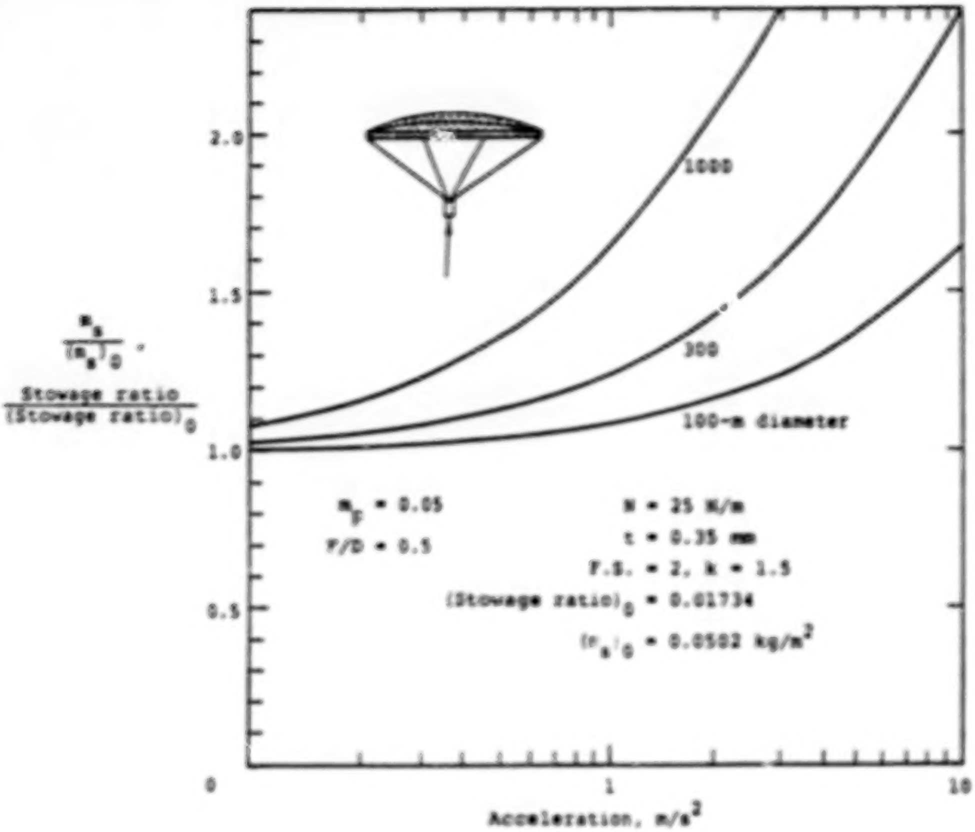


Figure 13. Mass and stowage ratio for geodesic dome.

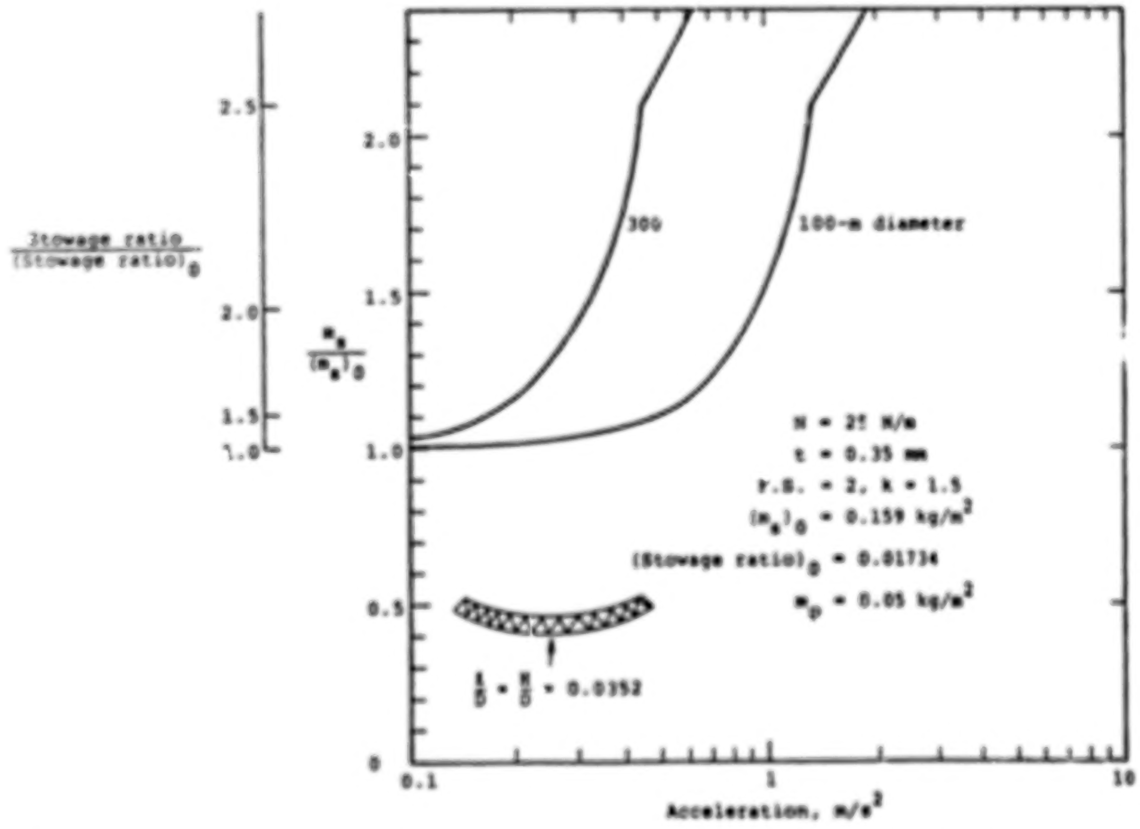


Figure 14. Mass and stowage ratio for tetrahedral truss.

INTEGRATED ANALYSIS OF LARGE SPACE SYSTEMS

Joseph P. Young
NASA Goddard Space Flight Center

Based on the belief that actual flight hardware development of large space systems will necessitate a formalized method of integrating the various engineering discipline analyses, an overall objective was established to produce an efficient highly user oriented software system capable of performing interdisciplinary design analyses with tolerable solution turnaround times. To support expected increase in large space systems design activities in the last half of the 1980's, a goal has been set to have a Version 1 IAC functioning by the end of FY 1983.

- OVERALL OBJECTIVE

  PRODUCE AN ANALYSIS SOFTWARE SYSTEM CAPABLE OF PERFORMING INTERDISCIPLINARY DESIGN ANALYSES OF LARGE SPACE SYSTEMS. MULTI-DISCIPLINES, WITH INITIAL EMPHASIS ON THERMAL, STRUCTURES, AND CONTROLS, ARE TO BE INTEGRATED INTO A HIGHLY USER ORIENTED ANALYSIS CAPABILITY. THE KEY FEATURE OF THE INTEGRATED ANALYSIS CAPABILITY IS TO BE AN EFFICIENT SYSTEM THAT WILL MINIMIZE SOLUTION TURNAROUND TIME.

- SPECIFIC NEAR TERM GOAL

  HAVE VERSION 1 OPERATIONAL INTEGRATED ANALYSIS CAPABILITY (IAC) FUNCTIONING BY END OF FY 1983.

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To be more definitive, specific analysis capability goals were set forth with initial emphasis given to sequential and quasi-static thermal/structural analysis and fully coupled structural/control system analysis. Subsequently, the IAC would be expanded to include a fully coupled thermal/structural/control system, electromagnetic radiation, and optical performance analyses.

ANALYSIS CAPABILITY

INITIAL EMPHASIS

- THERMAL/STRUCTURAL COUPLED ANALYSIS IN SEQUENTIAL MODE
- STRUCTURAL/CONTROL SYSTEM COUPLED ANALYSIS
- STRUCTURAL/CONTROL SYSTEM COUPLED ANALYSIS INCLUDING A PRIORI DEFINED TEMPERATURES (QUASI-STATIC THERMAL)

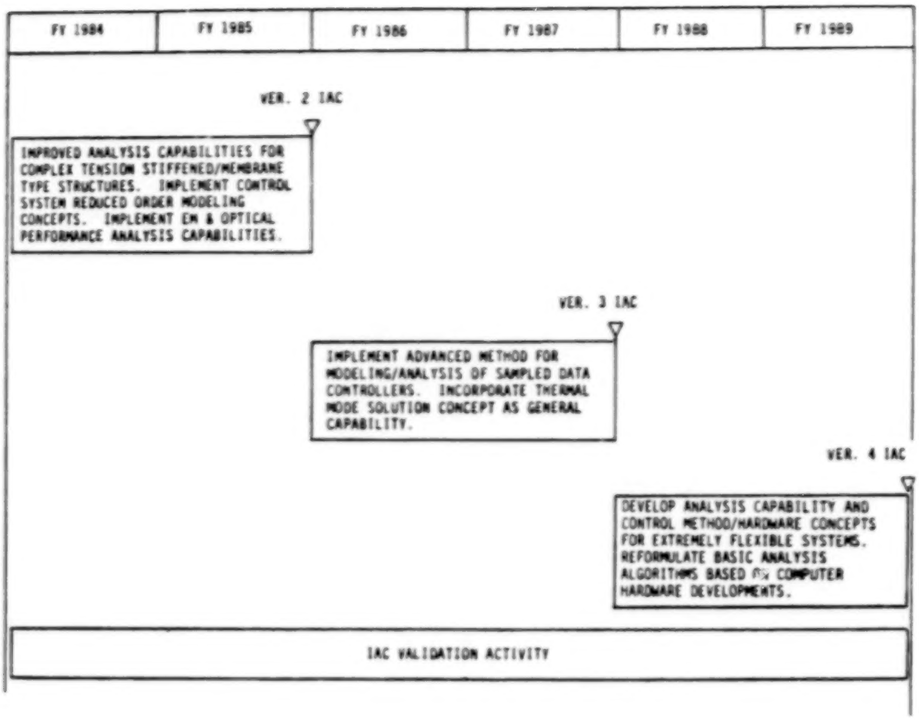
EXPANDABLE TO INCLUDE

- CLOSED LOOP THERMAL/STRUCTURAL/CONTROL SYSTEM ANALYSIS VIA USE OF THERMAL MODE CONCEPT
- ELECTROMAGNETIC RADIATION ANALYSIS
- OPTICAL PERFORMANCE ANALYSIS

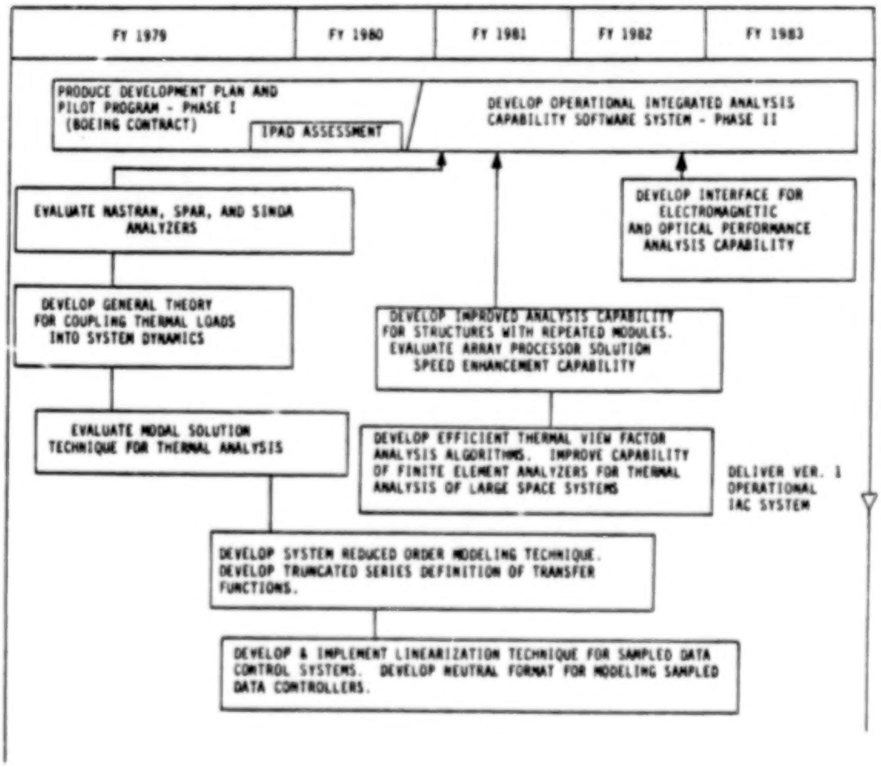
These two charts present a 10-year schedule that depicts a somewhat detailed picture of activities supporting the end of FY 1983 goal of a Version 1 IAC system and a general definition of tasks that support a delivery of enhanced versions of the IAC on 2-year intervals. The top bar in the first chart represents a key contract effort to produce the Version 1 IAC. Boeing Aerospace Company was awarded the Phase I portion July 1979. Completion is scheduled for July 1980. The contract contains a negotiated option to proceed with the Phase II operational software development/delivery portion. Underlying the major contract effort are a number of independent in-house activities at NASA centers that collectively provide support to the overall IAC development plan.

During the 6-year period following release of the Version 1 IAC, there is envisioned a progression of improved versions that will be upgraded to have capabilities for analyzing highly complex tension stiffened/membrane type structures, advanced method for modeling/analyzing sampled data control systems, and analyzing extremely flexible systems.

INTEGRATED ANALYSIS DEVELOPMENT PLAN


FY 1984
FY 1985
FY 1986
FY 1987
FY 1988
FY 1989
VER. 2 IAC
IMPROVED ANALYSIS CAPABILITIES FOR COMPLEX TENSION STIFFENED/MEMBRANE TYPE STRUCTURES. IMPLEMENT CONTROL SYSTEM REDUCED ORDER MODELING CONCEPTS. IMPLEMENT EM & OPTICAL PERFORMANCE ANALYSIS CAPABILITIES.
VER. 3 IAC
IMPLEMENT ADVANCED METHOD FOR MODELING/ANALYSIS OF SAMPLED DATA CONTROLLERS. INCORPORATE THERMAL MODE SOLUTION CONCEPT AS GENERAL CAPABILITY.
VER. 4 IAC
DEVELOP ANALYSIS CAPABILITY AND CONTROL METHOD/HARDWARE CONCEPTS FOR EXTREMELY FLEXIBLE SYSTEMS. REFORMULATE BASIC ANALYSIS ALGORITHMS BASED ON COMPUTER HARDWARE DEVELOPMENTS.
IAC VALIDATION ACTIVITY

INTEGRATED ANALYSIS DEVELOPMENT PLAN


FY 1979
FY 1980
FY 1981
FY 1982
FY 1983
PRODUCE DEVELOPMENT PLAN AND PILOT PROGRAM - PHASE I (BOEING CONTRACT)
IPAD ASSESSMENT
DEVELOP OPERATIONAL INTEGRATED ANALYSIS CAPABILITY SOFTWARE SYSTEM - PHASE II
EVALUATE NASTRAN, SPAR, AND SINDA ANALYZERS
DEVELOP INTERFACE FOR ELECTROMAGNETIC AND OPTICAL PERFORMANCE ANALYSIS CAPABILITY
DEVELOP GENERAL THEORY FOR COUPLING THERMAL LOADS INTO SYSTEM DYNAMICS
DEVELOP IMPROVED ANALYSIS CAPABILITY FOR STRUCTURES WITH REPEATED MODULES. EVALUATE ARRAY PROCESSOR SOLUTION SPEED ENHANCEMENT CAPABILITY
EVALUATE MODAL SOLUTION TECHNIQUE FOR THERMAL ANALYSIS
DEVELOP EFFICIENT THERMAL VIEW FACTOR ANALYSIS ALGORITHMS. IMPROVE CAPABILITY OF FINITE ELEMENT ANALYZERS FOR THERMAL ANALYSIS OF LARGE SPACE SYSTEMS
DELIVER VER. 1 OPERATIONAL IAC SYSTEM
DEVELOP SYSTEM REDUCED ORDER MODELING TECHNIQUE. DEVELOP TRUNCATED SERIES DEFINITION OF TRANSFER FUNCTIONS.
DEVELOP & IMPLEMENT LINEARIZATION TECHNIQUE FOR SAMPLED DATA CONTROL SYSTEMS. DEVELOP NEUTRAL FORMAT FOR MODELING SAMPLED DATA CONTROLLERS.

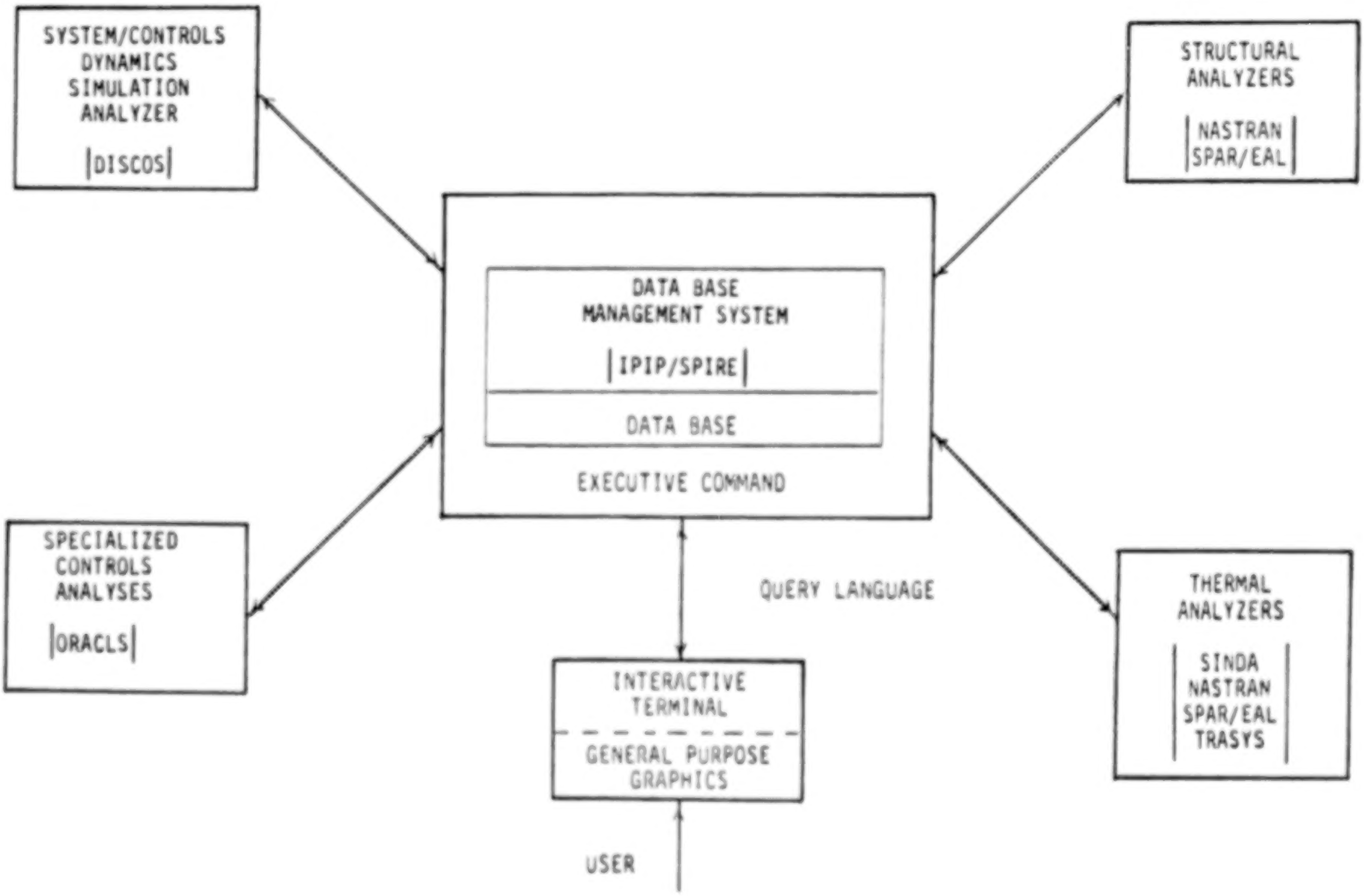
To give general guidance to this program, in the near term, there has evolved some Development Guidelines. The key motivation behind these guidelines is the objective to produce an efficient operational system within a minimum time frame and budget and that will have widest potential usage.

DEVELOPMENT GUIDELINES

- MAXIMIZE USE OF STATE-OF-ART TECHNOLOGY FOR ELEMENTS IN THE SYSTEM
- CONCENTRATE EFFORT ON THE TECHNOLOGY TO INTEGRATED SYSTEM ANALYSIS PROCESS INTO AN EFFICIENT TOOL
- TO PRODUCE SYSTEM WITH WIDEST POTENTIAL USEAGE, WITHIN MINIMUM BUDGET, INITIAL EMPHASIS GIVEN TO NEW BREED OF SUPER-MINICOMPUTERS AS HOST MACHINE
- UTILIZATION OF EXISTING DBMS IS PLANNED
  - BAC IS EVALUATING IPAD/IPIP. THIS IS PREFERRED APPROACH.
  - GSFC IS MODIFYING AN EXISTING DBMS FOR SPECIFIC PROJECT USEAGE. THIS SYSTEM (SPIRE) IS UNDER CONSIDERATION AS AN ALTERNATIVE APPROACH.

This diagram shows conceptually what the end product IAC is to be from a simplified architectural standpoint. The core of the IAC is a DBMS/Executive Command/Data Query capability. The individual technical discipline analyzers, illustrated by the surrounding blocks, are linked together through this central data manager/query system via data flow links (double arrows) thereby producing an Integrated Analysis Capability. These analyzers may exist external to the DBMS as implied by this diagram (i.e., interfaced with the DBMS), or one or more may, from a software standpoint, be integrated into the DBMS. Also shown are the current candidate codes that are seriously being considered for inclusion into the Version 1 IAC.

CONCEPT OF IAC ARCHITECTURE


SYSTEM/CONTROLS
DYNAMICS
SIMULATION
ANALYZER
DISCOS
STRUCTURAL
ANALYZERS
NASTRAN
SPAR/EAL
DATA BASE
MANAGEMENT SYSTEM
IPIP/SPIRE
DATA BASE
EXECUTIVE COMMAND
SPECIALIZED
CONTROLS
ANALYSES
ORACLS
QUERY LANGUAGE
THERMAL
ANALYZERS
SINDA
NASTRAN
SPAR/EAL
TRASYS
INTERACTIVE
TERMINAL
GENERAL PURPOSE
GRAPHICS
USER

NASTRAN and SPAR/EAL are considered the current premier general purpose structural analyzers. Over the past several years there have been conflicting opinions on the relative speed/efficiency of one vs. the other. It was believed to be worthwhile for a controlled comparison evaluation to be made, that is, where there is one person that understands the strong features of both codes, that uses the same computer, the same demonstration problems, and uses comparable versions of the codes.

CANDIDATE STRUCTURAL ANALYSIS CODES

CODES:

1. MSC 52 NASTRAN
2. COSMIC 17.5 NASTRAN
3. COSMIC SPAR
4. EAL

COMPARISON FACTORS:

- o SUITABILITY FOR USE IN INTERDISCIPLINARY ANALYSIS SYSTEM
- o LARGE PROBLEM ANALYSIS
- o EASE OF USEAGE
- o USER COMMUNITY
- o MAINTENANCE
- o DOCUMENTATION

This table gives a qualitative picture of the comparison showing, for example, that the MSC NASTRAN and SPAR are quite comparable on execution speed. Overall, the table currently shows MSC NASTRAN to be the most preferable although SPAR/EAL does show considerable potential. In terms of the IAC development, one result of this study has been to lead us to the decision to include both capabilities in the IAC.

STRUCTURAL CODES COMPARISON
PRELIMINARY OBSERVATIONS

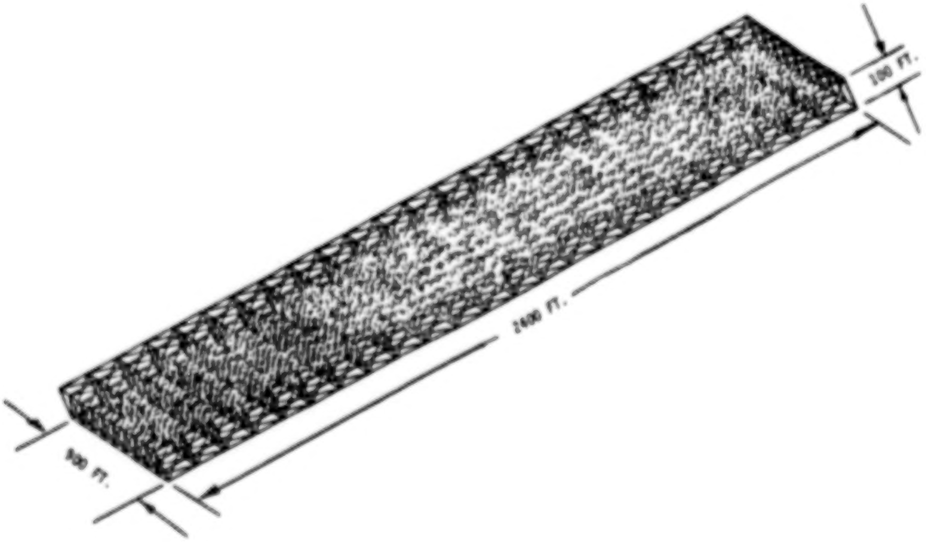
CODE	EXECUTION SPEED	EASE OF USAGE	DUCUMENTATION	USER COMMUNITY	MAINTENANCE
MCS 52 NASTRAN	+	-	+	+	+
COSMIC 17.5 NASTRAN	-	-	+	+	?
COSMIC SPAR	+	+	-	-	-
EAL	TDB	+	-	-	+

Another significant reason for conducting this study was to evaluate the performance applicability of the new breed superminicomputers to large space systems analyses. These two figures illustrate this study. The first figure gives representative minicomputer (DEC VAX 11/780) CPU run times for progressively larger size demonstration problems. The second figure depicts the type of demonstration problem utilized. A plate like structure serves as a good test model since it exhibits a relatively large bandwidth stiffness matrix thereby taxing the computing power of the host computer system. It must be emphasized that these times are only representative of the approximate times one might expect on a superminicomputer be it using either NASTRAN or SPAR. It is expected that about 6000 DOF will be the maximum possible dynamics solution problem due to exceeding typical external memory capacity limitations (25 Mb).

STRUCTURAL TEST PROBLEMS

REPRESENTATIVE MINICOMPUTER CPU RUN TIMES

MODEL	DOF	STATICS	DYNAMICS
A	48	–	0.5 MIN.
B	108	–	1 MIN.
C	1200	5 MIN.	25 MIN.
D	3000	15 MIN.	1 1/4 HR.



LSST PLATFORM-MODEL D

In the thermal analysis area, three aspects have been of concern.

- One, there has been a long standing question of finite element thermal analyzers vs. the finite difference modeling methodology and, in particular, as it pertains to radiation dominated thermal problems.
- Two, how best to compute heat flux input and needed thermal view factors.
- Three, understanding the possible utilization of thermal modes, as would reuslt from a classical eigenvalue analysis, in the world of large space systems thermal analysis.

THERMAL ANALYSIS EFFORT

- NASTRAN((NTA)-SINDA COMPARISON
- SELECTION OF RADIATION, FLUX/VIEWFACTOR, MODULE
- THERMAL MODAL ANALYSIS
  - COMPLETELY COUPLED ANALYSIS
  - REDUCTION OF THERMAL PROBLEM SIZE

The following five charts gives a picture of the IAC as viewed through the eyes of a controls system analyst/designer. Put very briefly, the objective is viewed as providing both a time and frequency domain analysis capability.

IAC-SIMULATION OF SYSTEM DYNAMICS

OBJECTIVE

PROVIDE AN INTERDISCIPLINARY ANALYSIS CAPABILITY
SUPPORTIVE OF BOTH TIME AND FREQUENCY DOMAIN
DESIGN AND PERFORMANCE EVALUATION METHODS

A focus problem that will exercise the IAC system to a very large degree is envisioned to be a Sampled Data Controlled Thermally Deformable Spacecraft. In addition, several functional types of control systems may be required. A focus problem of this type will lead to a number of analysis needs.

IAC-FOCUS PROBLEM

SAMPLED DATA CONTROL OF A THERMALLY DEFORMABLE SPACECRAFT

CONTROL SYSTEM TYPES:

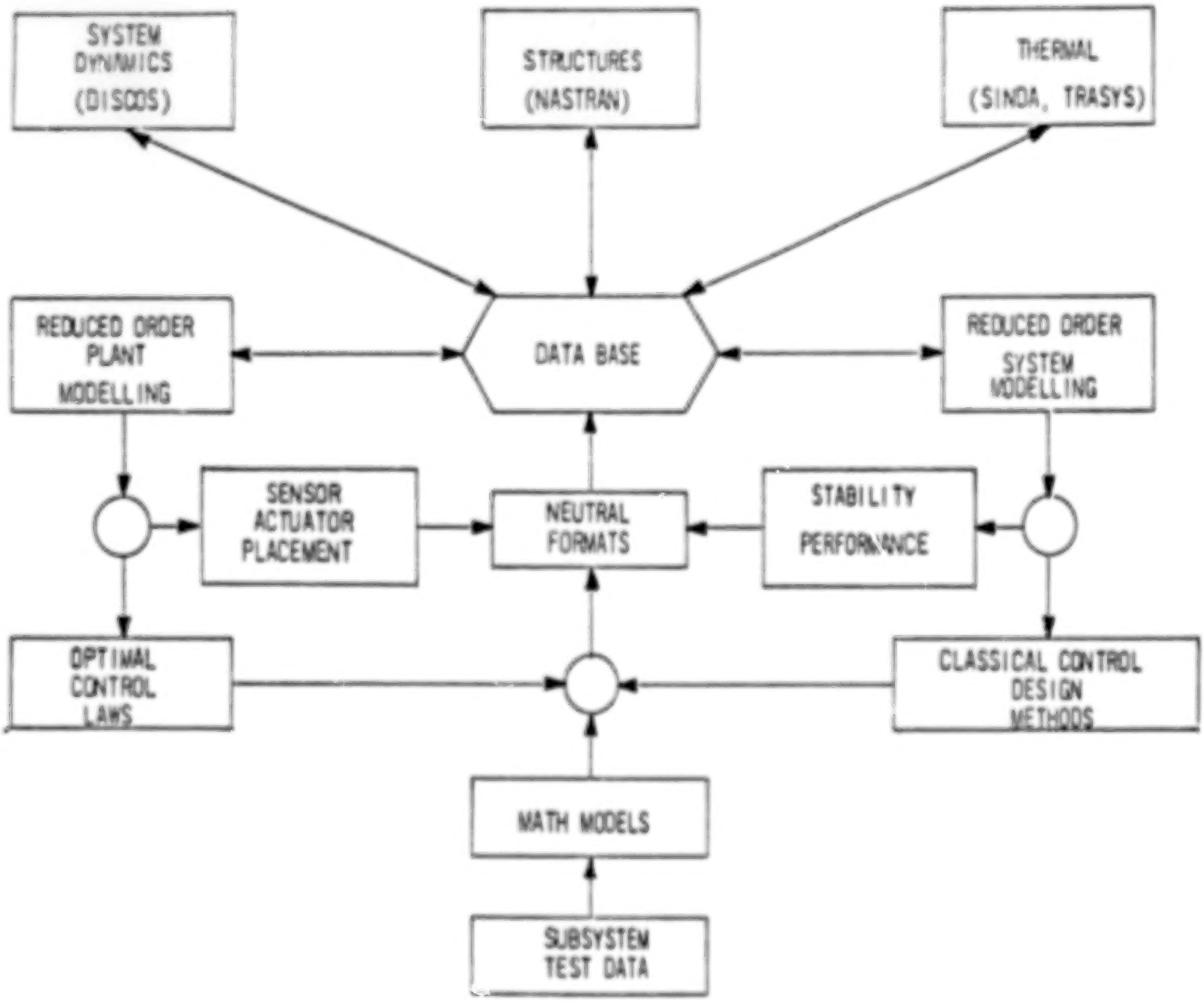
- SPACECRAFT ATTITUDE – POSITION – SHAPE CONTROL
- APPENDAGE POINTING CONTROL
- CONSTRUCTION AND DOCKING CONTROL

ANALYSIS TOOLS FOR THE DETERMINATION OF:

- LOADS AND DEFORMATION
- THERMAL RESPONSE
- SENSOR – ACTUATOR PLACEMENT
- OPTIMAL CONTROL LAWS
- FREQUENCY DOMAIN RESPONSE
- NON-LINEAR PERFORMANCE

This is another view of the IAC architecture with the control system analysis aspect expanded in greater detail showing the modern control theory contribution on the left and the classical control theory coming in on the right.

DATA FLOW PATHS FOR INTERDISCIPLINARY ANALYSIS



SYSTEM DYNAMICS (DISCOS)
STRUCTURES (NASTRAN)
THERMAL (SINDA, TRASYS)
REDUCED ORDER PLANT MODELLING
DATA BASE
REDUCED ORDER SYSTEM MODELLING
SENSOR ACTUATOR PLACEMENT
NEUTRAL FORMATS
STABILITY PERFORMANCE
OPTIMAL CONTROL LAWS
CLASSICAL CONTROL DESIGN METHODS
MATH MODELS
SUBSYSTEM TEST DATA

This chart contains a list of the most obvious environmental effects that must be considered indicating that capability to account for gravity gradient and thermal loads currently exists. No generalized capability exists for the remaining three loading sources. In addition, there exist some problems related to coupling the thermal and structures disciplines.

ENVIRONMENTAL EFFECTS

GRAVITY GRADIENT - CAPABILITY IN DISCOS
THERMAL - THERMAL INPUT INTO THERMAL ANALYZER VIA TRASYS
SOLAR PRESSURE
AERODYNAMIC DRAG
MAGNETIC
} NO GENERALIZED CAPABILITY

PROBLEMS:

- INTERPOLATION FOR THERMAL DEFORMATION
  THERMAL NODES TO STRUCTURAL GRID POINTS
- INTERPOLATION FOR THERMAL INPUT FOR CLOSED LOOP DYNAMICS
  GRID POINT STRUCTURAL DISPLACEMENT TO THERMAL SURFACE ORIENTATION(TRASYS)
- SOLAR PRESSURE AND AERODYNAMIC DRAG
  ADAPT TRASYS, ASSUME ONLY FREE MOLECULAR FLOW
- MAGNETICALLY INDUCED DEFORMATION DUE TO LARGE DIAMETER CURRENT CARRYING LOOPS

Modern and classical control theories will interface naturally in several areas of consideration and as a consequence produce a number of problems as shown in this figure.

MODERN-CLASSICAL CONTROL

SENSOR/ACTUATOR PLACEMENT
OPTIMAL CONTROL LAWS
FREQUENCY RESPONSE METHODS
NON-LINEAR PERFORMANCE

PROBLEMS:

- EFFECT OF SENSOR/ACTUATOR MASS ON PLANT DYNAMICS
- EFFECT OF SENSOR/ACTUATOR DYNAMICS IN CONTROL LAW IMPLEMENTATION
- OBTAIN ESTIMATOR MODEL FOR OPTIMAL CONTROL WORK
- OBTAIN LINEAR EQUATIONS-SAMPLED DATA CONTROL OF CONTINUOUS PLANT
- OBTAIN REDUCED ORDER SYSTEM EQUATIONS AND REDUCED ORDER TRANSFER FUNCTIONS
- DEVELOP EFFICIENT NUMERICAL INTEGRATION METHOD FOR MIXED STIFF (THERMAL)-OSCILLATORY (STRUCTURAL)-SAMPLED DATA (CONTROLLER) SYSTEM EQUATIONS

INTEGRATED ANALYSIS CAPABILITY
SUMMARY

- MAXIMIZE USE OF APPROPRIATE AVAILABLE ANALYZERS AND DBMS
- DEVELOP NECESSARY DATA FLOW LINK SOFTWARE TO BUILD IAC
- DEFINE FUNCTIONAL REQUIREMENTS TO BE SATISFIED BY ALL ELEMENTS OF THE IAC
- MODIFY "AS SUPPLIED" SOFTWARE ELEMENTS TO SATISFY FUNCTIONAL REQUIREMENTS
- DEVELOP IMPROVEMENTS TO BASIC ANALYZERS, TECHNIQUES FOR IMPROVING CONTROL SYSTEM MATH MODELING/ANALYSIS PROCESS, IMPROVED NUMERICAL SOLUTION ALGORITHMS, AND ANALYSIS SCHEMES FOR REDUCING DEMANDS ON COMPUTER HARDWARE DATA STORAGE CAPACITY
- DO SOFTWARE INTEGRATION TECHNOLOGY DEVELOPMENT NECESSARY TO MOLD ALL THE ELEMENTS INTO A USER FRIENDLY IAC WITH OBJECTIVE TO PROVIDE AN EFFECTIVE MEANS OF COMMUNICATING INTERDISCIPLINE DATA IN A TIMELY AND EFFICIENT MANNER

INTEGRATED ANALYSIS CAPABILITY
FOR LARGE SPACE SYSTEMS

Robert G. Vos
Boeing Aerospace Company

Program Objective

Develop "an integrated analysis computer program capable of performing the conceptual/preliminary structural system design analysis of large space systems in a highly efficient and rapid fashion."

Program Status

Contract NAS5-25767

Starting date: June 28, 1979

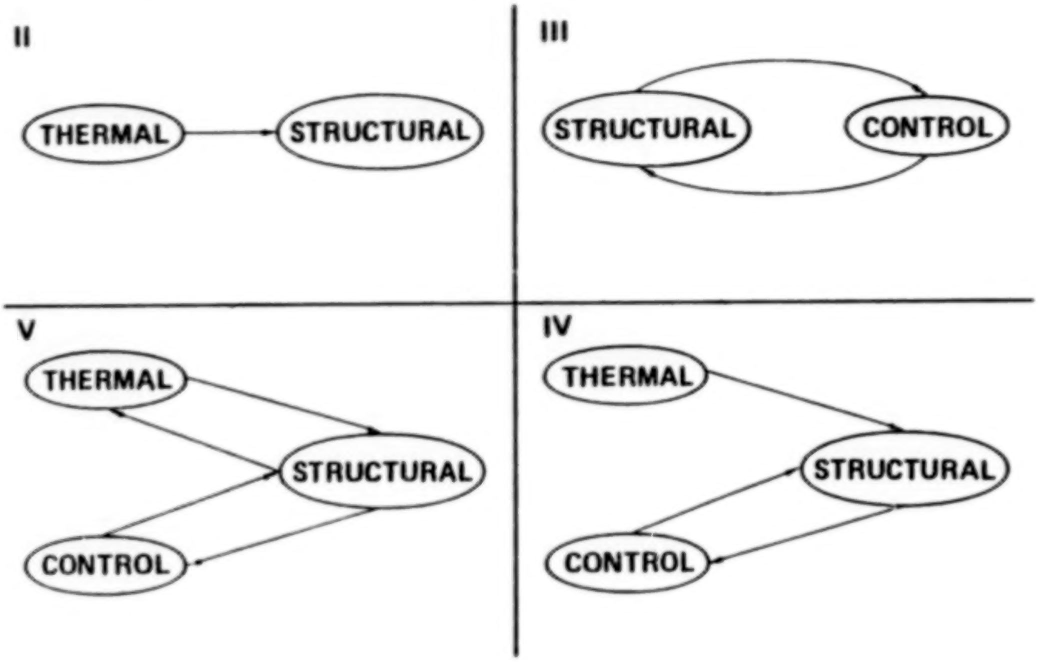
Duration of phase I: 10 months

Phase I:
- Task 1 – Generate a detailed development plan for the IAC
- Task 2 – Produce a simplified pilot analysis code

IAC Specifications

- LARGE SPACE STRUCTURES DESIGN ANALYSIS
- THERMAL/STRUCTURAL/CONTROLS INTEGRATION
- LATE–CONCEPTUAL/EARLY–FINAL DESIGN
- EMPHASIZE EXISTING SOFTWARE
- EMPHASIZE NON-PROPRIETARY SOFTWARE
- PROJECT SIZE
  - 1 TO 50 USERS
  - 1 TO 5 USERS CONCURRENT
- HOST COMPUTERS – "VAX-LIKE"
  - LARGE VIRTUAL MEMORY
  - HANDS-ON USER ENVIRONMENT
  - LOW COST ANALYSIS
  - MODERATE SIZE PROBLEMS (500 NODES)
- EMPHASIZE INTERACTIVE GRAPHICS AND I/O
- GROWTH POTENTIAL – EASY INCORPORATION OF NEW MODULES
- PROGRAMMING LANGUAGE – FORTRAN '77
- SCHEDULE – OPERATIONAL FY-83

IAC - CAPABLE OF PERFORMING


II
THERMAL
STRUCTURAL
III
STRUCTURAL
CONTROL
V
THERMAL
STRUCTURAL
CONTROL
IV
THERMAL
STRUCTURAL
CONTROL

IAC–Capable of Performing

- Thermal/structural analysis in a standalone mode
- Thermal/structural coupled analysis in a sequential mode
- Structural/control system coupled analysis
- Quasi-static thermal/structural/control system coupled analysis
- Fully coupled thermal/structural/control system analysis

INTERDISCIPLINARY DATA FLOW ITEMS

II - THERMAL/STRUCTURAL
- O THERMAL-LOADING (NODAL-TEMPERATURE) MATRIX
- O MODEL DESCRIPTION
- O MATERIAL DEFINITION

III - STRUCTURAL/CONTROL
- O NODE LOCATIONS
- O MASS PROPERTIES
- O STIFFNESS MATRIX
- O DAMPING MATRIX
- O MODE SHAPES
- O "A" AND "B" MATRICES
- O CONTROL ROUTINES

IV - THERMAL/STRUCTURAL/CONTROL (TRANSIENT)
- O ITEMS IN (II)
- O ITEMS IN (III)
- O THERMAL DEFORMATION (ELASTIC MODES)

V - THERMAL/STRUCTURAL/CONTROL (FREQUENCY DOMAIN)
- O ITEMS IN (III)
- O CAPACITANCE/CONDUCTANCE MATRICES
- O LINEARIZED RADIATION MATRIX
- O NEW "THERMAL MODE" TECHNOLOGY ROUTINES/MODULES

Technical Modules

SYSTEM DYNAMICS

DISCOS

THERMAL

MSC NASTRAN

COSMIC NASTRAN

SPAR

TRASYS

SINDA

STRUCTURAL

MSC NASTRAN

COSMIC NASTRAN

SPAR

CONTROLS

ORACLS

Structural/System Dynamics

COMPUTER PROGRAMS

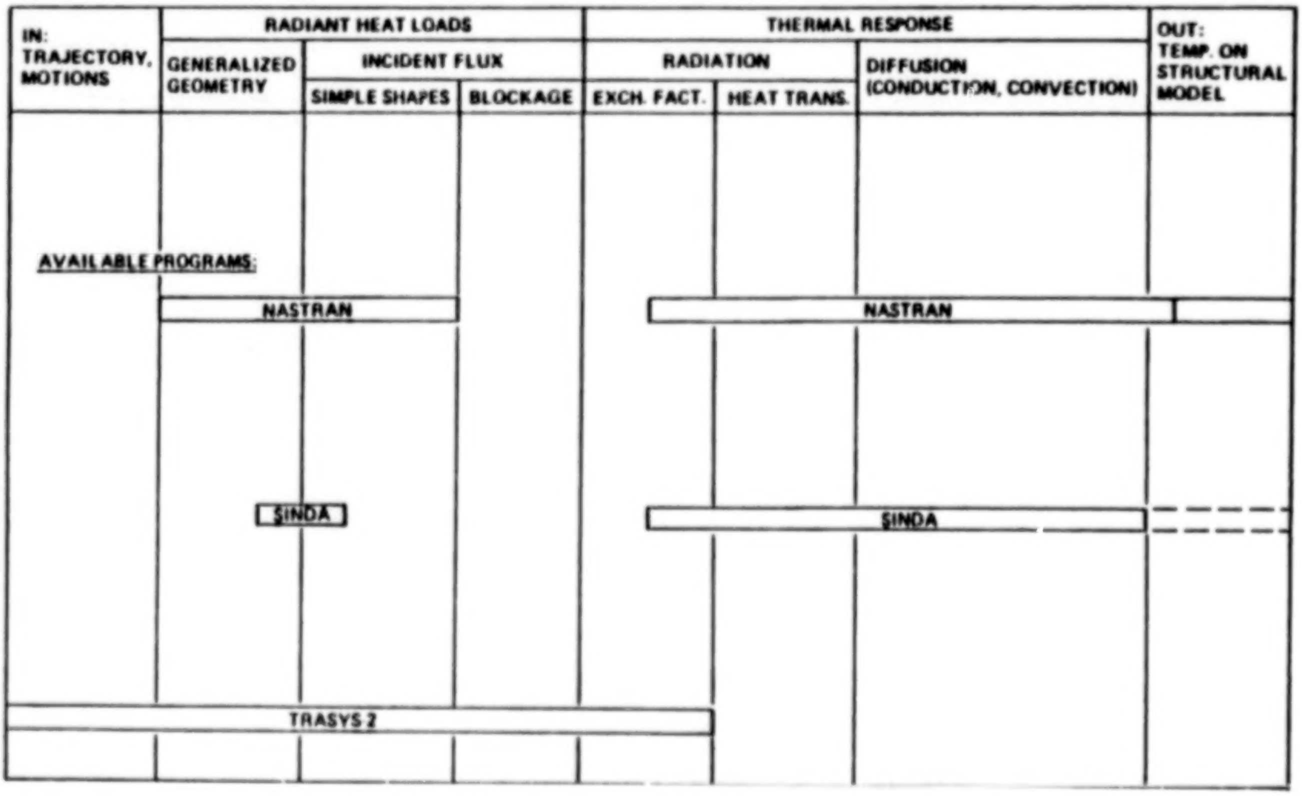
DISCOS - DYNAMIC INTERACTION SIMULATION OF CONTROLS AND STRUCTURE

- APPLICABLE FOR LARGE SPACE STRUCTURES
  - MULTI-BODY CAPABILITY
  - CONTROL SYSTEM/STRUCTURE INTERACTION
  - LARGE DISPLACEMENT (NONLINEAR) TIME DOMAIN ANALYSIS
  - LINEAR TIME AND FREQUENCY DOMAIN ANALYSIS
- USER CONVENIENCES
  - GRAPHICS OUTPUT
  - COMPUTER CODE MAINTAINED BY COSMIC
  - FLEXIBILITY FOR USER SUPPLIED SUBROUTINES

NASTRAN - NASA STRUCTURAL ANALYZER

- WIDELY USED AND AVAILABLE COMPUTER CODE
- MAINTAINED BY MACNEAL-SCHWENDLER CORP. (MSC) AND COSMIC
- MANY TYPES OF STRUCTURAL ELEMENTS AVAILABLE
- NASTRAN/DISCOS INTERFACE PROGRAM EXISTS

Thermal Programs


IN: TRAJECTORY, MOTIONS
RADIANT HEAT LOADS
GENERALIZED GEOMETRY
INCIDENT FLUX
SIMPLE SHAPES
BLOCKAGE
THERMAL RESPONSE
RADIATION
EXCH. FACT.
HEAT TRANS.
DIFFUSION (CONDUCTION, CONVECTION)
OUT: TEMP. ON STRUCTURAL MODEL
AVAILABLE PROGRAMS:
NASTRAN
NASTRAN
SINDA
SINDA
TRASYS 2

RECOMMENDATIONS FOR INCLUSION IN IAC

INCLUDE	DO NOT INCLUDE	TO BE DETERMINED
ORACLES	EASY5	DOPTSYS
	MDELTA	DIGIKON
	LAMP	MODEL
	CSAP	ROMP
	TAF	
	OPTSYS	

INTERDISCIPLINARY DATA FLOW

WHAT ARE THE BARRIERS?

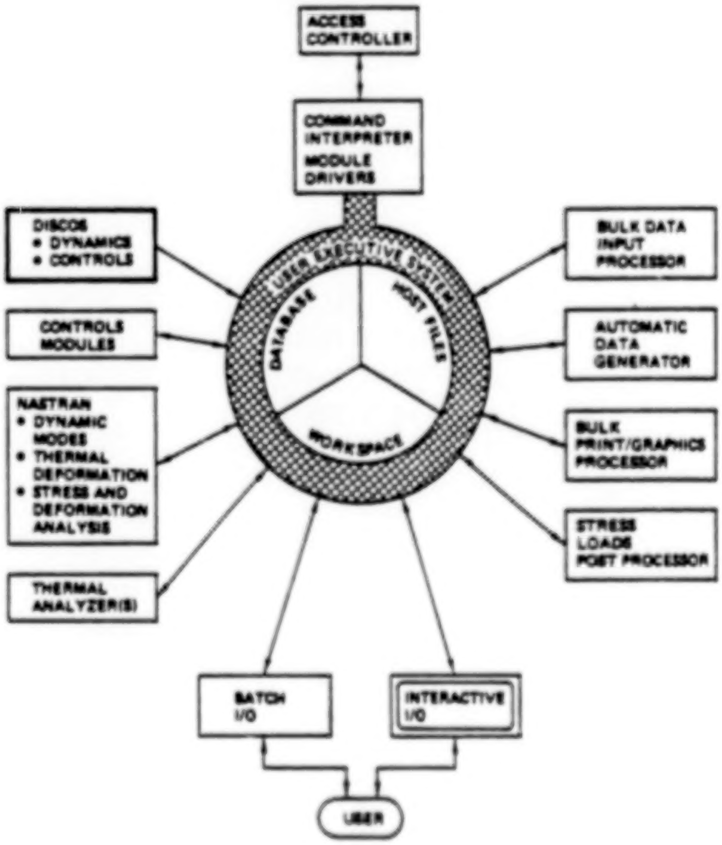
- POOR GROSS-LEVEL COMMUNICATION TOOLS
  - ANALYSIS ON DIFFERENT MACHINES
  - STORAGE IN SEPARATE FILES
  - NON-STANDARD DATA ACCESS
- SPECIFIC DATA-FLOW ITEMS ARE ILL-DEFINED
- DATA INCONSISTENCIES
  - O TECHNICAL FORM
    - FINITE-DIFFERENCE VS. FINITE ELEMENT MODELS
    - LUMPED VS. CONSISTENT MASS MATRICES
    - ABSOLUTE VS. RELATIVE ACCELERATION SOLUTIONS
    - BODY DEFINITIONS, TYPE OF CONNECTIONS
    - NUMBER AND TYPE OF MODES REPRESENTED
    - LEVEL OF DISCRETIZATION
    - ETC.
  - O STRUCTURE
    - UNSTRUCTURED FILES
    - SPARSE/FULL MATRICES
    - RELATIONS
    - RECORD STRUCTURES
  - O FORMAT
    - REAL VS. DOUBLE PRECISION
    - FORMATTED VS. BINARY
    - FIXED VS. FREE FIELD

INTERDISCIPLINARY DATA FLOW

HOW CAN BARRIERS BE OVERCOME?

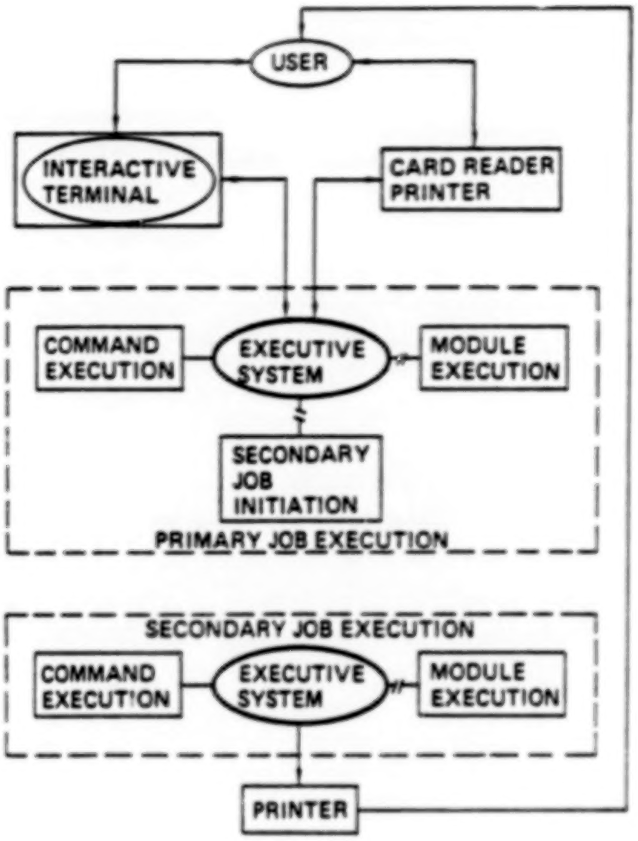
- PROVIDE A COMMON ANALYSIS SYSTEM AND DATABASE TOOL
- DEFINE AND ACCESS REQUIRED DATA-FLOW ITEMS
- OVERCOME DATA INCONSISTENCIES
  - O INTEGRATE THE TECHNOLOGIES
    - DESIRABLE WHERE POSSIBLE
    - SOME REQUIREMENTS INHERENTLY DIFFERENT
    - PREVENTS USE OF SOME EXISTING SOFTWARE
    - CHANGES UNACCEPTABLE TO SOME USERS
  - O ESTABLISH STANDARDS BUT PROVIDE INTERFACES
    - PREDEFINED FORMS/STRUCTURES/FORMATS
    - PERMANENT DATA STORAGE IS UNIQUE, NON-REDUNDANT
    - CONVERSIONS PERFORMED EACH TIME NEEDED
    - DEFINED BY SYSTEM SOFTWARE/MANAGER/USERS
    - IMPLEMENTED VIA MANDATE/GUIDELINE/COORDINATION
  - O USE DATA REDUNDANCY
    - PERMANENT STORAGE OF ALTERNATE DATA FORMS
    - EACH USER/TECHNOLOGY KEEPS OWN FORMS
    - CONVERSION REQUIRED WHEN DATA IS GENERATED/MODIFIED
    - REVERSIONS NOT ALWAYS POSSIBLE (ESSENTIAL FEATURES DESTROYED)

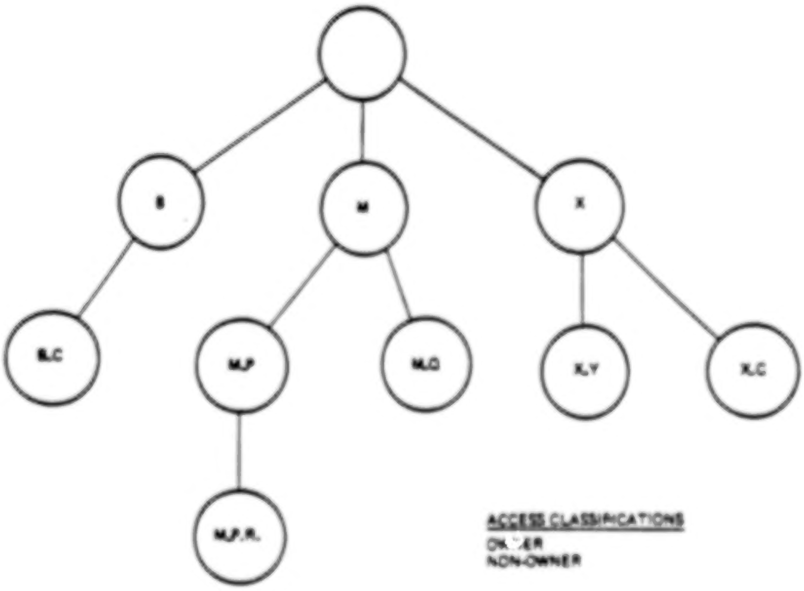
INTEGRATED ANALYSIS CAPABILITY FOR DESIGN OF LARGE SPACE SYSTEMS


ACCESS CONTROLLER
COMMAND INTERPRETER
MODULE DRIVERS
DISCOS
• DYNAMICS
• CONTROLS
CONTROLS MODULES
NASTRAN
• DYNAMIC MODES
• THERMAL DEFORMATION
• STRESS AND DEFORMATION ANALYSIS
THERMAL ANALYZER(S)
USER EXECUTIVE SYSTEM
DATABASE
HOST FILES
WORKSPACE
BULK DATA INPUT PROCESSOR
AUTOMATIC DATA GENERATOR
BULK PRINT/GRAPHICS PROCESSOR
STRESS LOADS POST PROCESSOR
BATCH I/O
INTERACTIVE I/O
USER

IAC Executive

COMMAND/MODULE/JOB FLOW


USER
INTERACTIVE TERMINAL
CARD READER PRINTER
COMMAND EXECUTION
EXECUTIVE SYSTEM
MODULE EXECUTION
SECONDARY JOB INITIATION
PRIMARY JOB EXECUTION
SECONDARY JOB EXECUTION
COMMAND EXECUTION
EXECUTIVE SYSTEM
MODULE EXECUTION
PRINTER


B
M
X
B,C
M,P
M,Q
X,Y
X,C
M,P,R.

ACCESS CLASSIFICATIONS
OWNER
NON-OWNER

ACCESS PRIVILEGES
PARTITION – CREATE, GATE
PARTITION INDEX – READ, DELETE
DATA STRUCTURE – READ, WRITE, DELETE

Database Logical Organization

IAC/IPIP Study

PURPOSE
  EVALUATE TECHNICAL FEATURES AND SOFTWARE OF IPIP FOR POSSIBLE USE IN THE IAC

GROUND RULES
  IPIP AVAILABLE ON VAX SYSTEM
  IPIP SOFTWARE OF OPERATIONAL QUALITY
  IPIP USER MODE (i.e., NO MODIFICATIONS OR MAINTENANCE)

EVALUATION CRITERIA
  TECHNICAL COMPATIBILITY WITH IAC REQUIREMENTS
  COMPATIBILITY WITH SHARED-USAGE VAX-LIKE ENVIRONMENT
  RISK/PORTABILITY/GROWTH
  IPIP/IAC DEVELOPMENT SCHEDULE
  IAC COST (DEVELOPMENT/OPERATIONAL/MAINTENANCE)

IAC Pilot Program

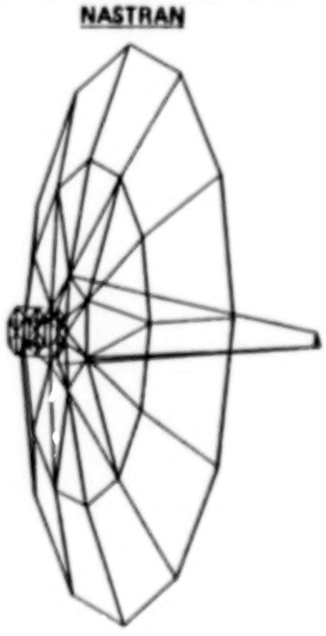
- TECHNICAL MODULES
  - DISCOS
  - MSC NASTRAN
- INTERFACE MODULES
  - NASTRAN THERMAL/STRUCTURAL
  - NASTRAN/DISCOS
- SOLUTION PATHS
  - THERMAL/STRUCTURAL
  - STRUCTURAL/CONTORLS
- EXECUTIVE
- DATABASE
- INTERACTIVE GRAPHICS
- VAX COMPUTER

Demonstration Problem

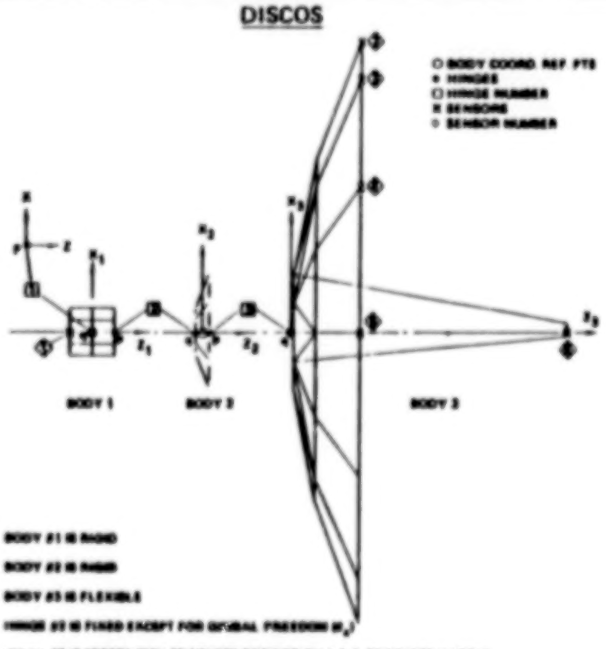
- 30-metre antenna
    - Bus
    - Reflector
    - Feed
- Solve two problem types:
    - Thermal/structural analysis
    - Structural/control analysis

IAC Demonstration Problem
Structural Models

NASTRAN



DISCOS


○ BODY COORD REF PTS
• HINGES
□ HINGE NUMBER
x SENSORS
○ SENSOR NUMBER
BODY 1
BODY 2
BODY 3

BODY #1 IS RIGID

BODY #2 IS RIGID

BODY #3 IS FLEXIBLE

HINGE #2 IS FIXED EXCEPT FOR GIMBAL FREEDOM (θ_2)

HINGE #3 IS "FREE" WITH STIFFNESS DEFINED BY A 6x6 STIFFNESS MATRIX

Work Breakdown Structure

1. MODULE INTERDISCIPLINARY DATA FLOW
2. EXECUTIVE SYSTEM
3. DATA HANDLING TOOLS
4. INTERACTIVE GRAPHICS
5. DEMONSTRATION AND ACCEPTANCE TESTING
6. MULTI-HOST COMPUTERS
7. DOCUMENTATION

CONTROLS FOR LSS

Fernando Tolivar
Jet Propulsion Laboratory

AGENDA

In this presentation we wish to summarize the various activities currently being carried out at JPL in the areas of control development for Large Space Structures. Secondly, we also wish to highlight some of the associated control problems.

The JPL activities are currently concentrated in 3 primary areas:

LSS MODELING
TECHNOLOGY IDENTIFICATION AND DEVELOPMENT
TECHNOLOGY APPLICATION AND PERFORMANCE EVALUATION

AGENDA

- CONTROL DEVELOPMENT OVERVIEW
- LSS MODELING FOR CONTROL SYNTHESIS
- TECHNOLOGY IDENTIFICATION AND DEVELOPMENT
- TECHNOLOGY APPLICATION AND PERFORMANCE EVALUATION

CONTROL DEVELOPMENT OVERVIEW

This viewgraph summarizes in a graphical form the interrelationships existing between the various elements of a controls development program for LSS.

A. LSS MODELING FOR CONTROL SYNTHESIS (Upper Left)

One of the areas that has been under intense investigation is that of modeling for controller design. This is widely recognized to be a major and, as yet, unresolved problem in achieving control of LSS.

B. TECHNOLOGY IDENTIFICATION AND DEVELOPMENT (Lower Left)

Another area of intense investigation is the identification and development of advanced control technology which will be required for the control of LSS. Substantial developments will be needed in the areas of distributed control, model order reduction/estimation, non-collocated sensors and actuators, static and dynamic shape control, etc.

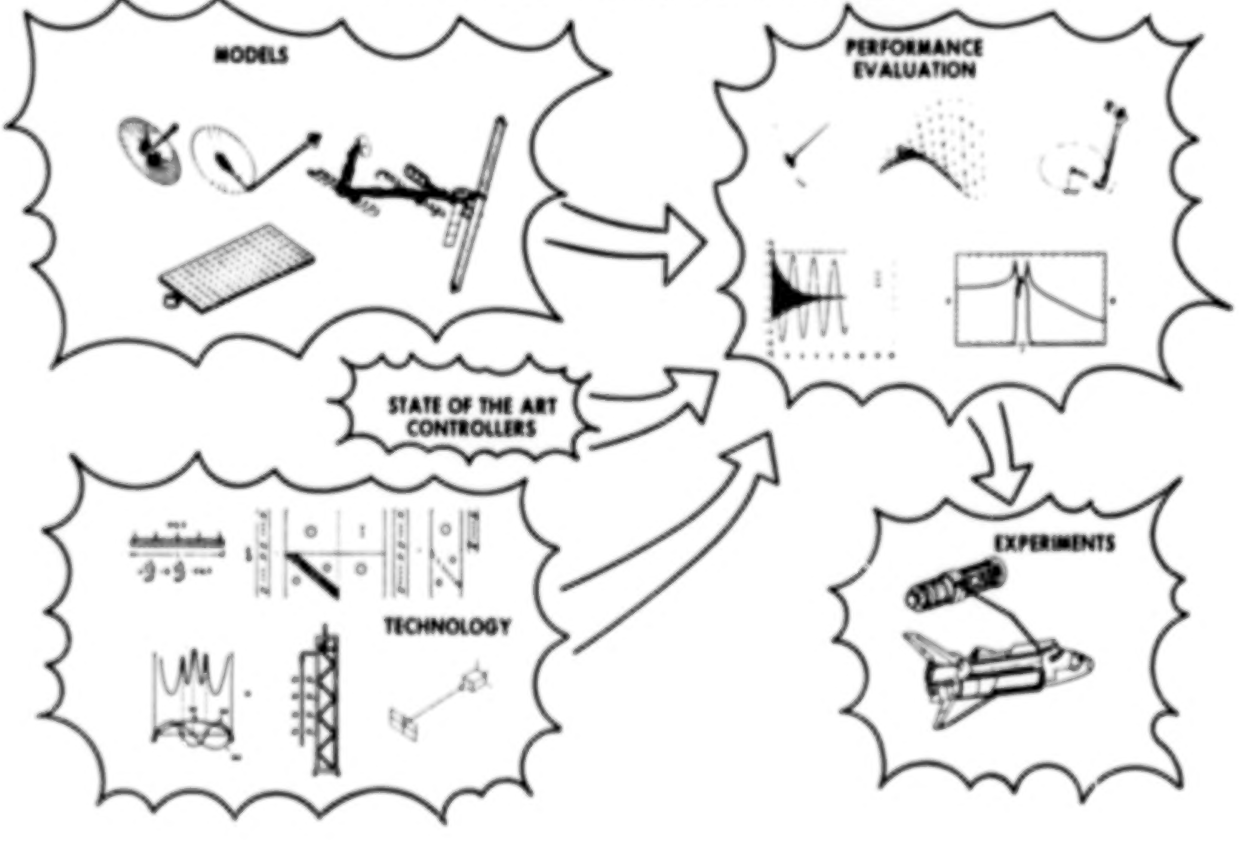
C. PERFORMANCE EVALUATION (Upper Right)

The performance afforded by current state-of-the-art control schemes as well as the limitations for their use in LSS is being assessed by means of simulations using the models developed under (A) areas found to be lacking feedback to (B) to drive the activities under TECHNOLOGY.

D. EXPERIMENTS (Lower Right)

Ultimately, the application or advanced control technology to LSS will have to be demonstrated by suitable flight experiments. Day-to-day developments will be validated through ground testing and laboratory experiments.

CONTROL DEVELOPMENT OVERVIEW



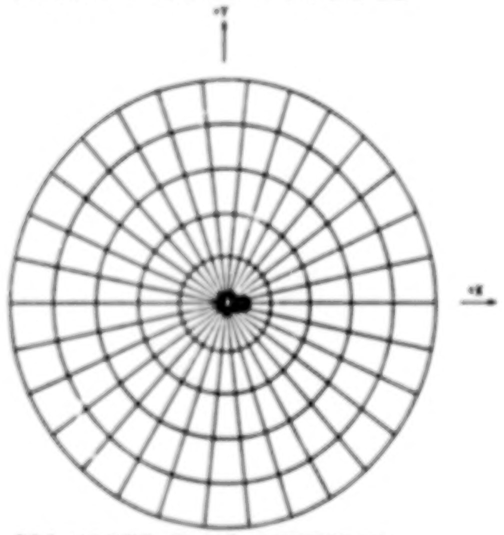
SYMMETRICALLY FED WRAP-RIB ANTENNA MODELS

A model for the 100-m diameter center-fed wrap-rib antenna has been developed to conduct attitude control and control/structure interactions analysis. The model consists of 184 nodes with 6 degrees of freedom per node. The natural modes vibration for the model have been computed and their characteristics are described on the table. The modeling activity is currently being extended to the offset feed configuration. Development of the offset feed design is required to reduce the problem of feed-support blockage.

SYMMETRICALLY FED WRAP RIB ANTENNA MODELS

- 100 meter DIAMETER
- 100 meter FEED SUPPORT
- 30 RIBS

FINITE ELEMENT MODEL



- 184 NODES PLUS REFERENCE
- 6 deg OF FREEDOM PER NODE
- 5 ELEMENT PER RIB
- GEOMETRIC STIFFNESS FOR MESH
- 4 NODES FOR BUS AND FEED

MODAL MODEL

MODE	FREQ, Hz	DESCRIPTION	CONTRIBUTING ERROR TYPE
1, 2, 3	0.00	RIGID-BODY TRANSLATION IN X, Y, Z DIRECTIONS	STATION CONTROL
4, 5, 6	0.00	RIGID-BODY ROTATIONS ABOUT X, Y, Z AXES	ATTITUDE/ POINTING CONTROL
7	0.053	REFLECTOR "UMBRELLA" MODE	DEFOCUS/GAIN
8, 9	0.065	FEED SUPPORT BENDING	POINTING
10, 11	0.073	REFLECTOR BENDING	SHAPE/GAIN
12, 13	0.094	REFLECTOR TORSION	DEFOCUS/GAIN
14, 15	0.096	REFLECTOR BENDING	SHAPE/GAIN
16, 17	0.118	REFLECTOR BENDING	SHAPE/GAIN
18, 19	0.140	REFLECTOR BENDING	SHAPE/GAIN
20	0.150	REFLECTOR TORSION	DEFOCUS/GAIN

- TO BE EXTENDED TO OFFSET FEED CONFIGURATION

MULTIPLE PAYLOAD SCIENCE APPLICATION PLATFORM (SASP) MODELS

Control technology developments are currently underway at JPL to address the attitude control of a SASP. A single axis 9 degree of freedom model has been completed. Transfer functions for this structural model have been found and controllability and observability of the 6 flexible modes has been determined. Emphasis of the study completed to date has been on obtaining a physical understanding of the parametric model developed and implications related to control system design.

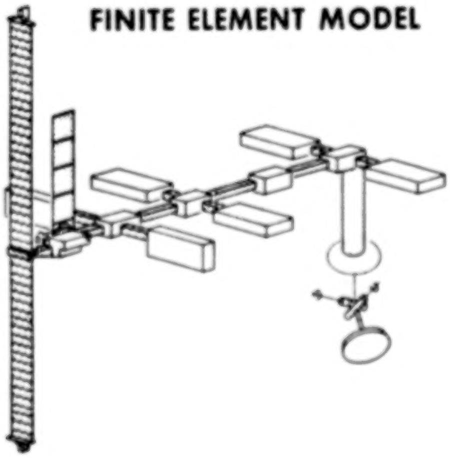
The left half of the viewgraph illustrates a fairly sophisticated structural configuration. The models developed to date consider only the solar panels, central bus structure, and the first two experiment modules (a T configuration).

The right half of the viewgraph illustrates a fairly sophisticated model for the solar panels. Such models have been developed for the solar electric propulsion vehicle and can be applied to the SASP if desired.

One of the most challenging aspects of the SASP is the interaction which results from several, possibly independent, control systems on board. Future studies will investigate the interactions of the experiment pointing control systems with the central bus control system.

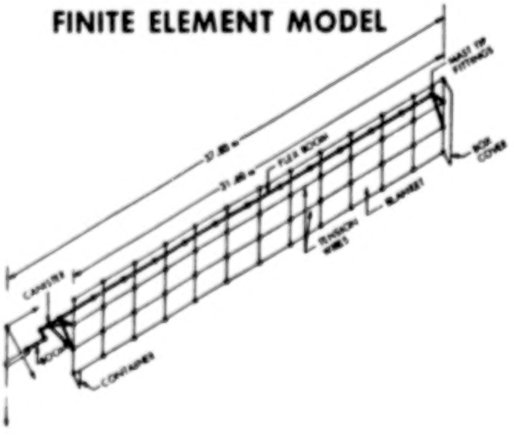
MULTIPLE PAYLOAD SCIENCE APPLICATION PLATFORM MODELS

PLATFORM FINITE ELEMENT MODEL



- SINGLE AXIS MODEL COMPLETED
- BEING EXPANDED TO INCLUDE ALL AXES AND FLEXIBLE EFFECTS
- ENABLES ANALYTICAL CONTROL/ STRUCTURE INTERACTION STUDIES

SOLAR PANEL FINITE ELEMENT MODEL



- 92 NODES
- 30 MODES RETAINED
- HINGE CONNECTED RIGID BODIES
- HYBRID COORDINATE SIMULATION
- TO BE INTEGRATED WITH PLATFORM MODEL

SOLAR POWER SATELLITE MODELS

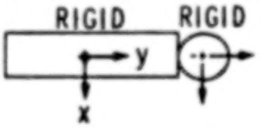
The SPS is the largest space system conceived to date that appears feasible with reasonable extensions of existing control technology. It represents a class of large platform-like structures that are several orders of magnitude larger than any of the other large space systems planned to date. The SPS has in common with all large space systems many control problems that are widely recognized within the controls community. The greatest need at the present time is to investigate the dynamics and control problems to assess performance of selected control concepts, and to identify and initiate development of advanced control technology that would enhance feasibility and performance of the SPS system.

One of the areas that has been under intense investigation is of modeling for controller design. This is widely recognized to be a major and as yet an unresolved problem in achieving precise control of large space systems. This problem arises because, to satisfy performance requirements, the control system must have the means for predicting very accurately the vehicle dynamic response. Yet, a precise large structure model is difficult to obtain because of the infinite degrees-of-freedom, nonlinearities, parameter uncertainty, difficulties in pre-flight dynamics testing, etc. This implies that the model in the control system design is at best a truncated approximation of the actual vehicle dynamics. A systematic selection of this approximate model is required.

Four distinct approaches have been developed in order to systematically select the controller design model. The models consist of a hinge-connected multibody model to conduct attitude dynamics and control studies, a continuum model to perform parametric studies of control/structure interaction dynamics, a complete flexible multibody model for performance prediction based on a comprehensive description of the vehicle dynamics, and a finite element model for the MPTS antenna for the study of structure deformation and prediction of scan losses due to local slope variations. Dynamic studies and parametric analysis using these models have revealed significant properties and provided insight to the dynamic behavior of the system. Our current emphasis is to apply these results to investigate the control problems.

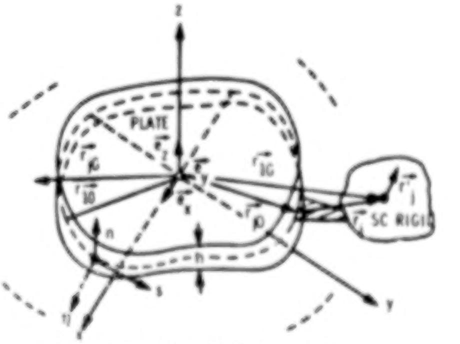
SOLAR POWER SATELLITE MODELS

MULTIBODY MODEL



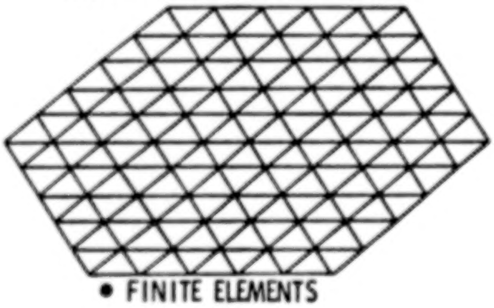
- 2 HINGE-CONNECTED BODIES
- INITIAL CONCEPT DEVELOPMENT

CONTINUUM MODEL



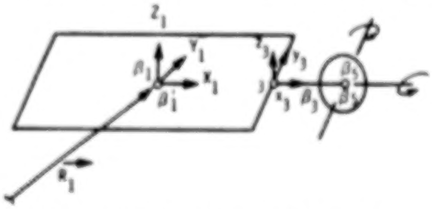
- 1 FLEXIBLE BODY WITH ATTACHMENTS
- PARAMETRIC ANALYSIS

MPTS ANTENNA MODEL



- FINITE ELEMENTS
- 167 NODES
- 20 MODES RETAINED

FLEXIBLE MULTIBODY MODEL



- 3 HINGE-CONNECTED BODIES
- COMPLETE DYNAMIC/CONTROL VERIFICATION

CONTROLS FOR LSS-TECHNOLOGY IDENTIFICATION AND DEVELOPMENT

As a result of the modeling activities for specific systems described in previous viewgraphs, a number of critical technology areas have been identified requiring further development. Current emphasis is in the areas of: 1) distributed control to achieve precise attitude and shape of large parabolic relfectors, 2) model order reduction required to find the best pre-flight dynamical models for controller design, 3) solution of the stability problems due to sensor and actuator noncolocation, 4) model error estimation for on-board detection and estimation of inevitable model errors such as parameter uncertainties, non-linearities, etc., 5) static and dynamic shape control necessary to remove structural biases due to thermal distortion, manufacturing tolerances, etc.

Future work will include: 1) development of adaptive estimation techniques required for on-board configuration of modeling deficiencies, 2) definition of sensing and actuation concepts for mechanization of distributed control in a representative application, and 3) laboratory demonstration of advanced concepts in a flexible-beam facility described in subsequent viewgraphs.

Space limitations preclude a detailed discussion of all of the foregoing areas. Emphasis will be focused in particular on unique approahces to the problem of shape control currently under investigation.

TECHNOLOGY IDENTIFICATION AND DEVELOPMENT

- CURRENT EMPHASIS
  - DISTRIBUTED CONTROL
  - MODEL ORDER REDUCTION
  - NON-COLOCATED SENSORS AND ACTUATORS
  - MODEL ERROR ESTIMATION
  - ⇒ STATIC AND DYNAMIC SHAPE CONTROL
- FUTURE WORK
  - ADAPTIVE ESTIMATION TECHNIQUES
  - SENSING AND ACTUATING CONCEPTS FOR DISTRIBUTED CONTROL
  - LABORATORY DEMONSTRATION OF ADVANCED CONTROL CONCEPTS

STATIC SHAPE CONTROL

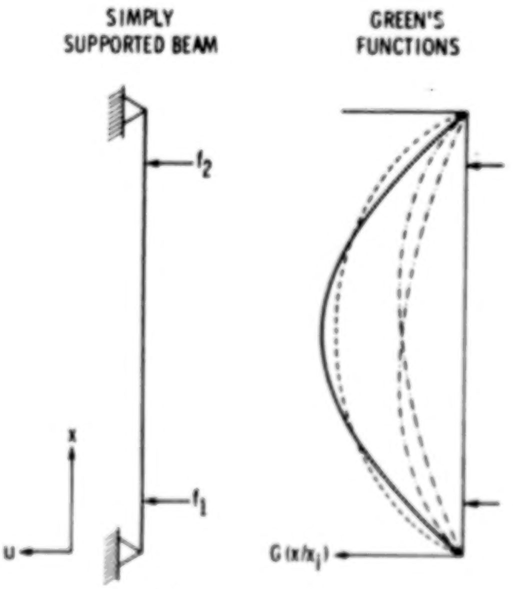
Lightweight flexible space structures are being designed which will exhibit dynamic shape variations greater than those of any previous spacecraft. The technology for providing the shape control necessary for adequate performance of these structures remains to be developed.

At JPL an approach using the Green's function, or influence coefficient, is being developed for representative system models. Shape control can be achieved by actuators placed at point locations along the structure. A system model consisting of a partial differential equation representing the change in shape, the right side of which represents a sum of forces f_i applied at the positions x_i, is displayed on the viewgraph.

The Green's function represents the response of the structure to a force of magnitude 1 at one point. Thus the total response (shape) of the structure is merely the sum of the Green's functions multiplied by the forces at each point. The viewgraph displays the visual interpretation of this fact for two forces.

As an example of this approach, suppose we wish to achieve some desired, say, parabolic shape U(x) by means of two forces applied as shown in the figure. The objective is to find the magnitude of the forces f_1 and f_2 which result in the best approximation to U(x) in the mean square sense. The solution to this problem is easily obtained by replacing the shape by its expression in terms of the unknown forces and the Green's functions. Standard minimization techniques can then be applied to obtain the optimum forces.

STATIC SHAPE CONTROL



$$\frac{\partial u^4}{\partial x^4} + \Omega \cancel{\frac{\partial^2 u}{\partial t^2}}^{\,0} = \sum_i \delta(x-x_i) \cdot f(x_i)$$

$$\Longrightarrow G(x/x_i)$$

ERROR AND EFFORT
MINIMIZATION

$$E(F) = \int_0^{\ell} \left|U(x) - u(x)\right|^2 dx + \cancel{q\sum_i f_i^2}^{\,0}$$

WHERE:

$$u(x) = \sum_i G(x/x_i) f_i$$

U(x) = DESIRED SHAPE

SET:

$$\frac{dE(F)}{dF} = 0$$

SOLVE FOR:

$$F = \begin{vmatrix} f_1 \\ f_2 \end{vmatrix}$$

GREEN'S FUNCTIONS - · - · - · -
ACHIEVED SHAPE - - - - - - -
DESIRED SHAPE ———

SHAPE CONTROL - GREEN'S FUNCTION APPLICATION CONSIDERATIONS

While Green's function techniques apply to linear system models, non-linear models may be accommodated by solving successive iterations using linearized models.

In addition to the ease with which the Green's function handles a combination of continuous and discrete (pointwise) functions, and enables constrained optimization problems to be solved, the theory provides readily computed approximate solutions to any desired accuracy through the use of eigenfunction (modal) expansions. The approach possesses nearly limitless practical and theoretical advantages.

GREEN'S FUNCTION APPLICATION CONSIDERATIONS

- INFLUENCE COEFFICIENTS FOR ARBITRARY ACTUATOR PLACEMENT CAN BE DETERMINED
- BASED ON LINEAR MODEL APPROXIMATION (SMALL DISPLACEMENTS)
- NON-LINEAR RANGE ACCOMODATED THROUGH ITERATION
- PROVIDES TECHNIQUE TO EVALUATE EFFECTIVENESS OF SURFACE ACTUATION SCHEMES

PERFORMANCE EVALUATION

The performance of advanced control concepts as well as conventional state-of-the-art controllers is being assessed by means of computer simulation using the structural models described earlier. This work is being carried out in three primary areas:

100 Meter Wrap Rib Antenna
Multiple Payload Science Application Payload
Solar Power Satellite

In addition, the need for actual laboratory verification of advanced control concepts has been identified and an experimental facility has been built for that purpose. The facility will permit verification of advanced control technology in the areas of vibration suppression, shape control, distributed control, adaptive control, non-collocated sensors and actuators, etc.

PERFORMANCE EVALUATION

- ⟹ 100 MTR WRAP RIB ANTENNA
  - SYMMETRICALLY FED
  - OFFSET FED
- MULTIPLE PAYLOAD SCIENCE APPLICATION PLATFORM
- SOLAR POWER SATELLITE
- ⟹ FLEXIBLE BEAM SHAPE CONTROL LABORATORY EXPERIMENT

PERFORMANCE EVALUATION - SYMMETRICALLY FED WRAP-RIB ANTENNA

A substantial portion of the control technology developments currently underway at JPL pertains to the attitude control of a 100-m wrap-rib antenna. Past efforts have addressed the symetrically fed antenna configuration and have resulted in the definition of 3 controller designs, the development of computer programs for simulation of the combined control/structure dynamics, and the generation of surface performance estimates for the attitude control design.

Recent efforts are being focused on the offset-fed antenna configuration discussed later.

PERFORMANCE EVALUATION

- SYMMETRICALLY FED WRAP RIB ANTENNA
    - ALTERNATE LUMPED CONTROLLERS DEFINED
        - PROPORTIONAL + DERIVATIVE
        - PROPORTIONAL + DERIVATIVE + INTEGERAL
        - OPTIMAL CONTROL DESIGN-MULTIPLE CRITERIA
    - LARGE SCALE CONTROL / STRUCTURE DYNAMIC SIMULATION
        - DISCRETE TIME-MODAL COORDINATES
        - 14 VIBRATIONAL MODES PLUS RIGID BODY RESPONSE
        - 3 DIMENSIONAL DYNAMIC DISPLAY
        - POINTING AND SURFACE DISTORTION COMPUTATION
    - COMPUTATION OF RF PARAMETERS BEING INCORPORATED

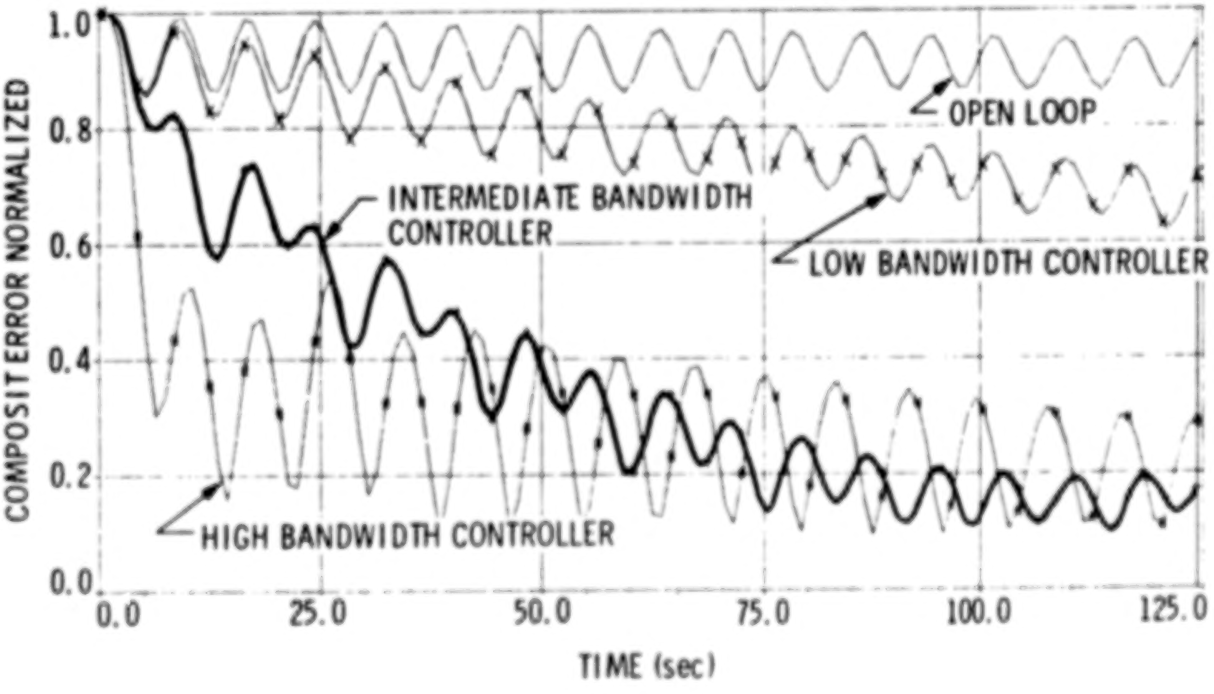
PERFORMANCE EVALUATION - SIMULATION RESULTS FOR SYMMETRICALLY FED 100-M ANTENNA

The performance of various attitude control designs has been investigated by means of computer simulations. The viewgraph shows a summary of this investigation. The quantity plotted on the vertical scale is proportional to the potential energy in the system and provides therefore a global, composite indication of the vehicle dynamic response. An initial excitation results in a lightly damped oscillatory open-loop response. Damping of 0.5% has been assumed for the simulation. The chart also displays the performance of three distinct types of controllers: (1) a "slow" controller with a low bandwidth, (2) an intermediate bandwidth system, and (3) a high-bandwidth or "fast" controller. It is of interest to note that the intermediate controller appears to perform better than both the fast and slow controllers. This result violates the intuitive notion that "slow" controllers are better because they provide for frequency separation between the controller bandwidth and the first natural frequency of the structure. Such results are to be expected because of the large number of modes and the highly interactive characteristics of the structure. Large structures do not always obey "rules of thumb" used in previous attitude control designs.

More important than the sample results displayed on the viewgraph is the development of the simulation capability itself. This simulation is currently being applied to determine the dynamic and control response of the offset-feed configuration.

PERFORMANCE EVALUATION
SIMULATION RESULTS

100 M ANTENNA PROPORTIONAL + DERIVATIVE + INTEGRAL CONTROLLER



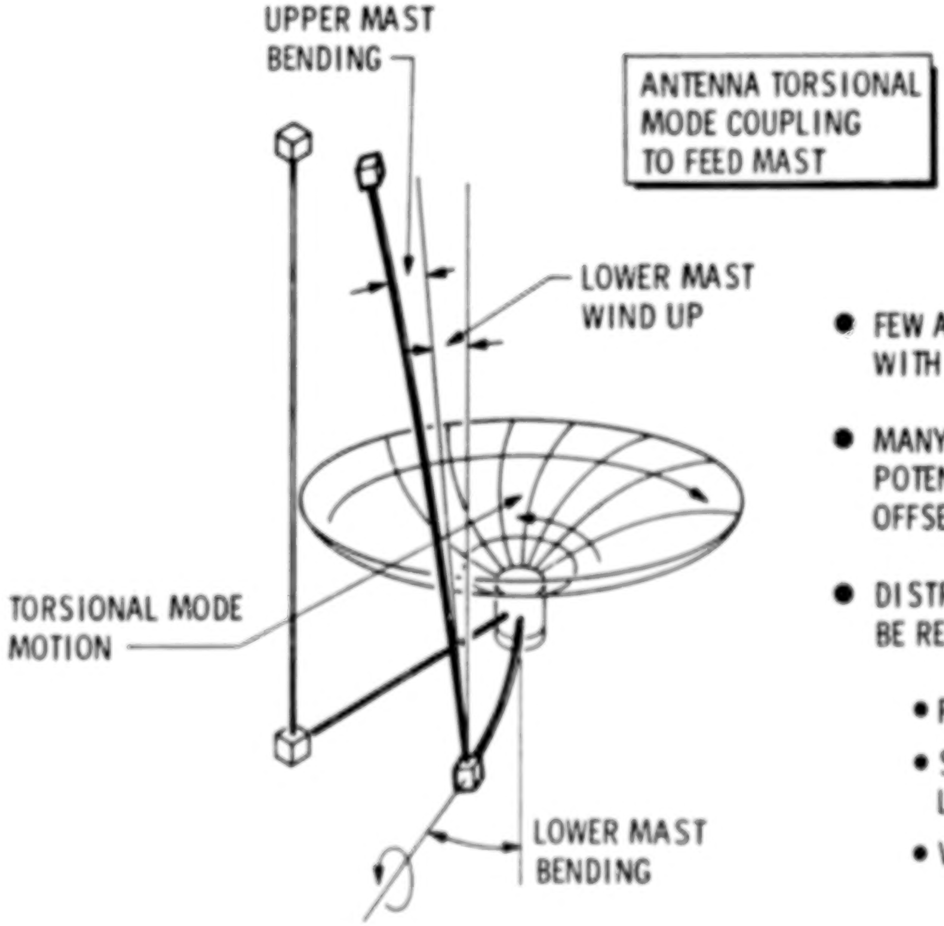
PERFORMANCE EVALUATION - OFFSET FED WRAP-RIB ANTENNA

A substantial portion of the control technology developments currently under way at JPL address the attitude control of a 100-m wrap-rib antenna configuration. The effort has resulted in the definition of preliminary attitude control designs, development of computer programs for simulation of the combined control/structure dynamics, and the generation of surface performance estimates for the attitude control design. Recent emphasis has been placed on an offset feed structure, although a center-fed antenna has also been studied in the past. Potential coupling between dish and feed mast modes makes the offset feed configuration a more challenging vehicle for control system design.

The viewgraph illustrates a total vehicle mode of the combined feed and dish components of the structure. In this mode, bending of the vertical upper mast couples with a combined bending/tension mode of the lower mast which in turn results in dish distortions. Such coupling means that the attitude control designs must account for the combined effects of overall vehicle attitude, motion of the feed with respect to the dish, and distortions of the surface shape. Additional complications arise in the offset design because of the uncertainties in the mass center location and cross products of inertia due to the lack of symmetry in the configuration.

No control technology currently exists that would guarantee successful control and inflight performance of such highly interactive systems. Substantial developments are required in the areas of distributed control, precision pointing, shape and feed location control in order to reduce the risk of attempting to fly these systems without the required control technology developments.

PERFORMANCE EVALUATION
OFFSET FED WRAP RIB ANTENNA



- FEW ANTENNA MODES COUPLE WITH SYMMETRICAL FEED MAST
- MANY ANTENNA MODES HAVE POTENTIAL FOR COUPLING TO OFFSET FEED MAST
- DISTRIBUTED CONTROL WILL BE REQUIRED FOR:
  - POINTING
  - SHAPE AND FEED LOCATION CONTROL
  - VIBRATION SUPPRESSION

PERFORMANCE EVALUATION - LABORATORY VERIFICATION

Large structures are infinite-dimensional systems that cannot be characterized fully by any model. Consequently, the controller design models will suffer from inevitable deficiencies due to truncated modes, parameter uncertainties, neglected nonlinearities and external disturbances. Such inevitable model errors will result in degraded performance and even overall unstable system behavior. The problem is not insoluble as approaches are currently under development that would guarantee satisfactory performance even in the presence of the modeling errors. However, substantial control technology developments must be carried out in the areas of distributed control, adaptive systems, and model order reduction in order to guarantee satisfactory overall system performance. Ultimately, the application of such control technology to LSS will have to be demonstrated by suitable flight experiments. However, the day-to-day developments will have to be validated through ground testing and laboratory experiments.

This viewgraph shows a photograph of one such laboratory experimental facility developed at JPL. The experiment consists of a hanging pinned-free 12-1/2 foot long stainless steel beam (6" wide, 1/32" thick). This configuration results in modal frequencies of 0.30, 0.74, 1.32, 2.00, 3.22, 5.72... hertz, and easily observed mode shapes. Four non-contacting eddy current position sensors and three brushless d.c. motor force actuators may be mounted at any station along the length of the beam. A microprocessor controller implements the estimation and control algorithms by sampling the sensors, updating the state estimates, and outputting the control command. The sample rate for a six state controller is twenty hertz.

PERFORMANCE EVALUATION

LABORATORY VERIFICATION

- VIBRATION SUPPRESSION
- STATIC SHAPE CONTROL
- DISTRIBUTED CONTROL
- ADAPTIVE CONTROL
- NON COLOCATED SENSORS AND ACTUATORS

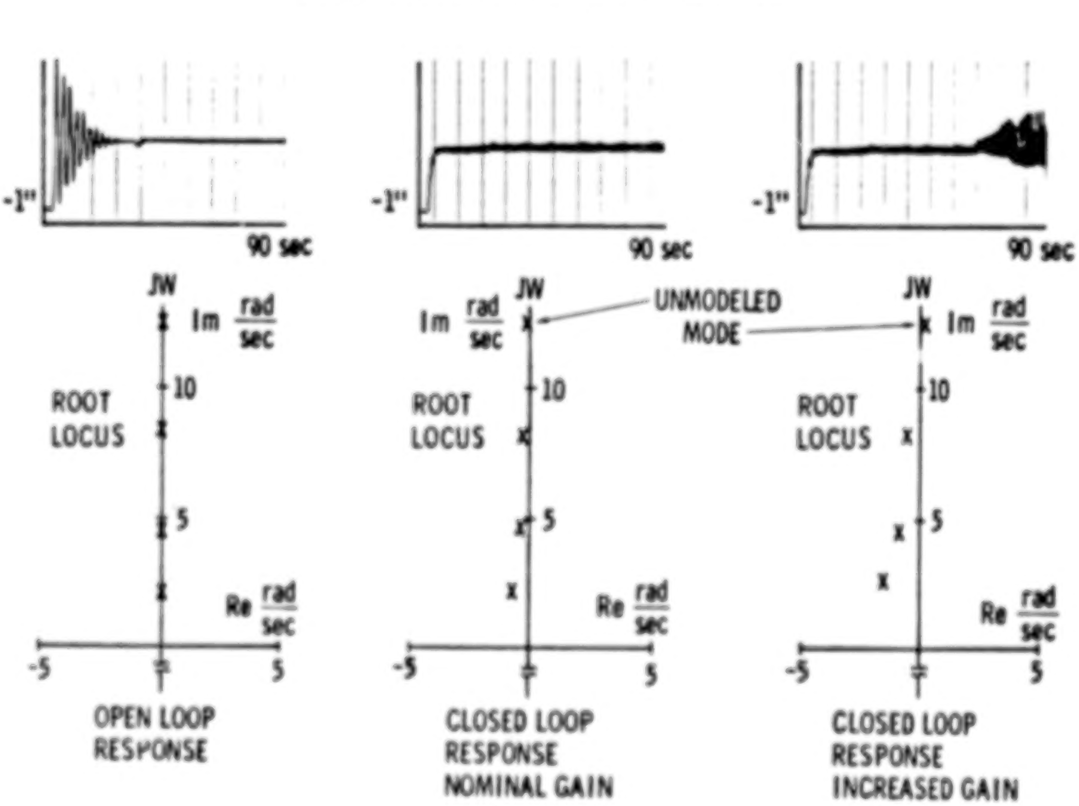
PERFORMANCE EVALUATION - BEAM EXPERIMENT

A major and as yet unsolved problem that will occur in the offset feed system is that of modeling for controller design. In order to achieve precise attitude and vibration control, the control system must have the means for predicting very accurately the vehicle dynamic response. For instance, a sufficiently precise model must be available to predict the feed-mast/dish interactions described earlier in order for the control system to reduce the resulting degradations in vehicle performance. However, paradoxically, such models are currently nonexistent for the offset-feed system and, in fact, will not become available in-flight until dynamical testing is carried out.

The viewgraph shows a concise statement of the modeling problem that is common to all large space systems including the offset-feed antenna. The viewgraph displays the response of a 12-1/2 foot flexible-beam experimental facility developed at JPL to verify control technology developments. The chart on the left corresponds to the response of the structure under an initial excitation. The response is governed primarily by a total of four natural modes of vibration. To illustrate the problems due to model truncation the control system design was based on the first three system modes without the inclusion of the fourth vibrational mode. The performance of the system is illustrated in the center of the viewgraph. The system very quickly reduces the initial excitation. However, as a result of the mode that was left out of the controller design, the system exhibits a residual oscillation that persisted throughout the duration of the experiment. The chart on the right of the viewgraph shows an even more unstable behavior due to an increase in the control system gain. The message left by this experiment is that degraded performance (as shown on the second chart) and even instabilities (as shown in the third chart) can and indeed do arise as a result of inaccuracies in the control system dynamical models.

While the hardware verification experiment has been performed on a flexible-beam model and not on the antenna systems, the results are generically applicable to both cases. A precise dynamical model for the antenna system will not be available as a result of any pre-flight analysis.

PERFORMANCE EVALUATION



CONCLUSIONS AND OBSERVATIONS

A summary has been presented of the various activities being carried out at JPL in the area of control development for Large Space Structures. From the foregoing, the following conclusions/observations can be made.

- No control technology currently exists that guarantees successful control and in-flight performance of highly interactive, flexible, large structures. A vigorous development effort in control technology is essential in order to reduce the high risk factor if we were to fly such systems without the necessary control technology development.

- New technologies should be validated, as far as possible, with ground testing/experiments design to minimize the effects of the ground environment. Larger structures not amenable to ground testing will require flight testing to characterize their dynamics and control/dynamic interactions. Such testing will be essential until control technology is sufficiently advanced to provide controllers which are insensitive or adaptive to dynamic uncertainties.

- The challenges of large structures bridge across traditional divisision by disciplines such as Controls, Mechanisms, Propulsion, Structures, Temperature Control, etc. The challenges are such that only an integrated design approach encompassing all these disciplines will enable future Large Space Systems.

CONCLUSIONS/OBSERVATIONS

- LSS INTRODUCE NEW CLASSES OF CONTROL TECHNOLOGY REQUIREMENTS AND A VIGOROUS DEVELOPMENT EFFORT IS ESSENTIAL

- NEW TECHNIQUES SHOULD BE VALIDATED WITH LABORATORY EXPERIMENTS DESIGNED TO MINIMIZE THE EFFECTS OF THE GROUND ENVIRONMENT

- LSS WILL HAVE SIGNIFICANT DYNAMIC UNCERTAINTY DUE TO MODELING ERRORS AND THE UNTESTIBILITY OF THESE SYSTEMS IN THE GROUND ENVIRONMENT

- FLIGHT TESTS WILL BE REQUIRED TO ADEQUATELY CHARACTERIZE THE DYNAMICS UNTIL CONTROL TECHNOLOGY PROVIDES CONTROLLERS INSENSITIVE OR ADAPTIVE TO DYNAMIC UNCERTAINTIES

- ONLY AN INTEGRATED CONTROL/STRUCTURE/MISSION DESIGN APPROACH WILL ENABLE FUTURE LARGE SPACE SYSTEMS

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ELECTRIC PROPULSION AND POWER

David C. Byers
NASA Lewis Research Center

Electric propulsion programs are in progress in Europe, Japan, the USA, and the USSR. About a half dozen space tests of electric propulsion have been performed by the USA and the USSR has published results of over a dozen space experiments. In the near future many space tests of electric propulsion are firmly planned by Japan (pulsed plasma, MPD, ion thruster); West Germany (ion thruster); and the USA (pulsed plasma, ion thruster).

Due to time constraints it is impossible to present aspects of all ongoing electric propulsion programs and for brevity only the NASA electric propulsion program will be discussed herein.

ELECTRIC PROPULSION PROGRAM

OBJECTIVE

- IDENTIFY, PROVIDE, AND TRANSFER THE TECHNOLOGY FOR ELECTRIC PROPULSION SYSTEMS FOR ON-ORBIT AND TRANSPORTATION PROPULSION FOR EARTH-ORBITAL AND PLANETARY MISSIONS

Electric propulsion converts electrical energy into directed momentum or fields which can be used for propulsion functions.

Electric propulsion offers the benefits of operation at values of specific impulse an order of magnitude or more greater than theoretically possible with chemical propulsion. This feature grossly reduces the propellant requirements for transportation and on-orbit propulsion functions which can result in enabled mission capability or significant reductions in mission costs. To date, emphasis has been on space propulsion devices. Recently, however, some efforts have been directed at electric propulsion concepts to augment Earth-to-orbit propulsion.

ELECTRIC PROPULSION SYSTEMS

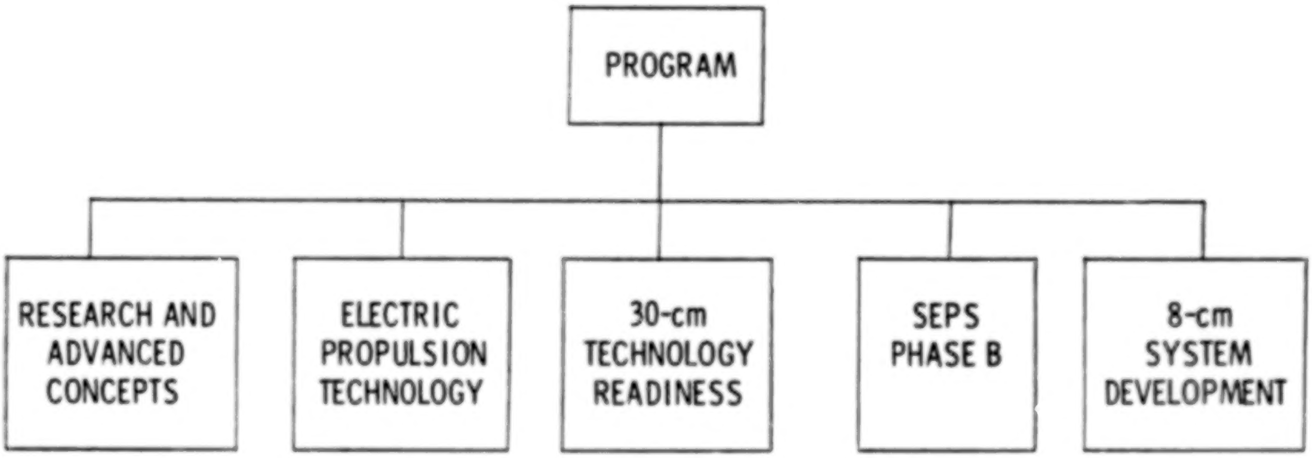
FUNCTION

TRANSFORM ELECTRICAL ENERGY
INTO DIRECTED MOMENTUM OR FORCE
FOR ON-ORBIT OR TRANSPORTATION PROPULSION FUNCTIONS

The NASA electric propulsion program encompasses R&D efforts on several concepts and range from basic research to final development and flight test.

The research and advanced concept efforts are presented in a subsequent discussion. A brief summary of the status of the various elements of the NASA electric propulsion program will be given on the following charts.

ELECTRIC PROPULSION PROGRAM



PROGRAM

- RESEARCH AND ADVANCED CONCEPTS
- ELECTRIC PROPULSION TECHNOLOGY
- 30-cm TECHNOLOGY READINESS
- SEPS PHASE B
- 8-cm SYSTEM DEVELOPMENT

ELECTRIC PROPULSION PROGRAM

- ENCOMPASSES:
    - R&D EFFORTS ON SEVERAL CONCEPTS
    - EFFORTS FROM BASIC RESEARCH TO FLIGHT TESTS
- ION THRUSTER SYSTEMS ARE THE MOST MATURE EP CONCEPT IN THE USA

Two 8cm mercury ion thrusters will be flown on the Air Force P80-1 satellite which will be launched from the Shuttle into a 740Km altitude polar orbit. One thruster will be placed on the zenith side and the other on a surface which is alternately the ram or wake side. The zenith thruster will demonstrate the propulsion functions required for seven years north-south stationkeeping of a 1000 Kg geosynchronous satellite. The thrusters will be operated simultaneously and in various modes to duplicate conditions expected on an operational system. Diagnostics are arranged about each thruster to refine ground based data on the particle effluents from the 8cm thrusters.

Successful culmination of this space test should provide adequate confidence in the hardware to allow for user application of the 8cm ion thruster system.

ION THRUSTER SYSTEMS

8-cm MERCURY

PROPULSION FUNCTION	CHARACTERISTICS	STATUS
• ON-ORBIT FOR EARTH ORBITAL MISSIONS	• ONE MLB • 175 W • 2800 SEC.	• IN FINAL DEVELOPMENT FOR SPACE TEST ON AF P80-1 SATELLITE • FLIGHT HARDWARE IN FAB. PHASE

NASA has been engaged for several years in a program to provide technology readiness of the 30cm mercury thruster system by the end in 1980. The 30cm thruster system was developed primarily for planetary transportation. In the technology readiness effort the thruster has been developed and its lifetime verified by a series of long term tests. The field and particle interfaces of the thruster are also being defined. Other critical technology, such as power conditioning circuits and elements, are also under development and their basic design will be verified in tests with thrusters.

Recently two Phase B system studies were initiated in industry to define Solar Electric Propulsion Systems (SEPS) capable of a number of missions. It is anticipated that these studies will result in overall SEPS approaches and provide sufficient definition to allow initiation of a final development program for SEPS.

ION THRUSTER SYSTEMS

BASELINE 30-cm MERCURY

PROPULSION FUNCTION	CHARACTERISTICS	STATUS
• TRANSPORTATION FOR PLANETARY MISSIONS	• 8-30 MLB • 0.75-3 kW • 2200-3000 sec.	• CRITICAL SYSTEM TECHNOLOGY READINESS TO BE ACHIEVED IN 1980 • PHASE B SYSTEM STUDIES UNDERWAY (MANAGED BY MSFC)

Advanced mercury thruster systems are under development for transportation and on-orbit propulsion for Earth orbital missions and transportation for planetary Nuclear Electric Propulsion Systems. For these applications, increase in thrust and thrust to power ratio provide strong performance and cost benefits. In addition, due to the nearly constant power, strong simplifications can be made in power processing. Tests are underway which indicate long lifetimes are available at increased thrusts and that significant reductions (~ 3X) in thrust system specific mass, and power to thrust (~ 2X) ratio are possible with advanced mercury ion thruster systems.

ION THRUSTER SYSTEMS

ADVANCED MERCURY

PROPULSION FUNCTION	CHARACTERISTICS	STATUS
• TRANSPORTATION FOR PLANETARY AND EARTH ORBITAL MISSIONS	• INCREASED THRUST & THRUST/POWER • SIMPLIFIED PPU REQS.	• THRUSTS TO ~ 0.1 LB DEMONSTRATED • SPECIFIC IMPULSES DOWN TO ~ 1500 sec. DEMONSTRATED • 500 HOUR LIFE TEST PERFORMED AT ~ 50 MLB • REDUCED POWER PROCESSOR REQS. DEMONSTRATED
• ON-ORBIT PROPULSION	• INCREASED THRUST & THRUST/POWER • SIMPLIFIED PPU & COMMAND/CONTROL REQS.	• THRUSTS TO 4 MLB DEMONSTRATED

Inert gas thruster systems are of interest for Earth orbital missions for several reasons:

1) The fact that the inert gases do not condense offers some strong performance benefits. These include the ability to start up the thruster system in a few seconds and the possibility of eliminating many power supplies;
2) The integration of thruster systems will become an increasingly important issue as the Earth orbital space systems increase in size and complexity. Inert gases are more benign than any other candidate propellants which should ease the integration of propulsion systems with the space systems;
3) Inert gases, due to their light atomic masses, inherently operate at higher values of specific impulse than mercury. Future Earth orbital missions are likely to include heavier space systems, last longer, and include more on-board power than present systems. All of these traits strongly drive propulsion systems in the direction of increased specific impulse;
4) For Earth orbital mission models which include many large space systems, the availability and potential environmental impact of mercury will probably preclude its use as a transportation, or perhaps on-orbit, propellant.

ION THRUSTER SYSTEMS

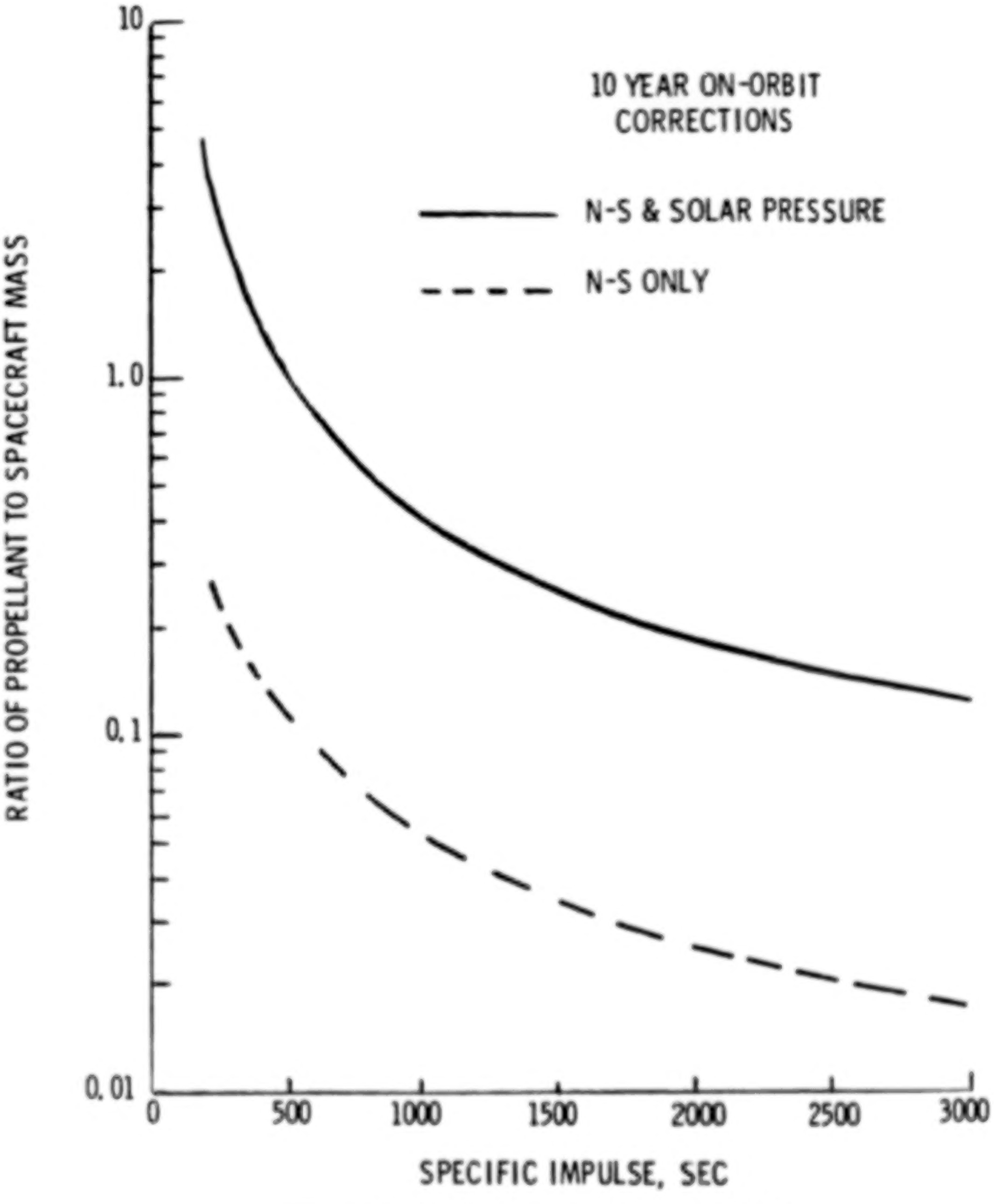
INERT GAS

PROPULSION FUNCTION	CHARACTERISTICS	STATUS
• TRANSPORTATION AND ON ORBIT FOR EARTH-ORBITAL MISSIONS	TBD	• RESEARCH PROGRAM IN PROGRESS FOR 4 YRS • PROGRAM ENTERING PRELIMINARY DEVELOPMENT PHASE

APPLICATIONS

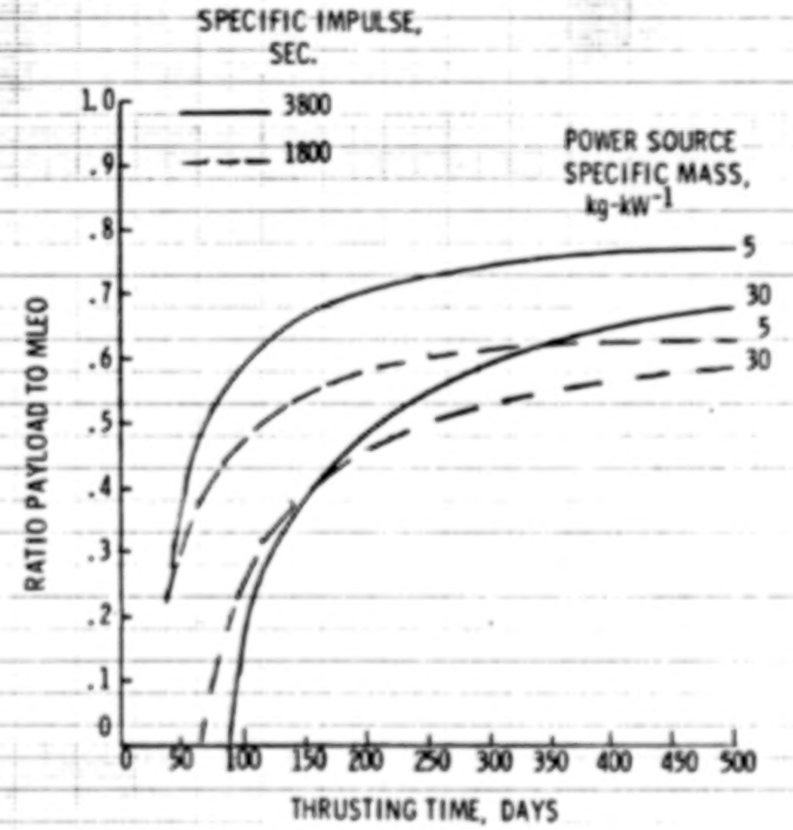
- PROPULSION REQUIREMENTS FOR LSS MAY DIVERGE SHARPLY FROM PRIOR EXPERIENCE
    - GREATLY INCREASED ON-ORBIT & TRANSPORTATION PROPULSION ENERGIES
    - NEW ON-ORBIT & TRANSPORTATION PROPULSION REQUIREMENTS
    - NEW MISSION STRATEGIES

This figure shows the ratio of propellant, required for geosynchronous on-orbit propulsion, to spacecraft mass as a function of specific impulse. The dotted curve is appropriate for dense spacecraft typical of those in use today. The effect of solar pressure increases directly with the ratio of system surface area to mass and that ratio is expected to be very much higher for future LSS than for present systems. The solid curve shows the propellant to mass ratio for a geosynchronous satellite with the characteristics of the Space Based Radar. It is seen that for systems with lightweight structure the on-orbit propellant requirements become very large and can exceed by factors the spacecraft mass for specific impulses less than about 500 seconds.



EFFECT OF SOLAR PRESSURE ON ON-ORBIT PROPULSION (SBR GEOMETRY)

The figure shows the ratio of non-power payload to total mass required in Low Earth Orbit (LEO) for a geosynchronous orbit (GEO) transfer using state-of-art mercury ion thrusters. In addition to the non-power payload, the electric propulsion thrust system and the power system are also delivered to GEO and are available for various uses on-orbit. Dependent on the specific impulse and specific power source increases, it is seen that GEO transfers are possible in less than 50 days. The non-power payload rises rapidly with trip time from a zero value to an asymptotic value dependent only on the specific impulse in the limit of very long trip times. The figure also shows that the non-power payload can become a large fraction of the total mass required in LEO. This feature can grossly reduce the Earth to orbit propulsion requirements for LSS at GEO.



APPLICATIONS

- EARLY DEFINITION OF GENERIC LSS PROPULSION REQUIREMENTS CAN ALLOW APPROPRIATE RESPONSES IN ELECTRIC PROPULSION TECHNOLOGY DIRECTIONS

ELECTRIC PROPULSION FOR SPS

Earle M. Crum
NASA Lyndon B. Johnson Space Center

SPS TRANSPORTATION REQUIREMENT:

- CONSTRUCT TWO 50,984 MT SPS's IN GEO PER YEAR, PLUS SPARES
- ANNUAL CARGO REQUIREMENT:
  - SATELLITE: 101,968
  - SPARES (1%): 1,020

102,988 MT

LEO TO GEO OTV SYSTEMS CONSIDERED

- LO_2/LH_2
  - PERMITS GEO CONSTRUCTION
  - SHORT TRIP TIME
  - SPS SOLAR ARRAY PROTECTED FROM VAN ALLEN RADIATION

  UP TO $1B PENALTY PER 5 GW SPS DUE TO GREATER PROPELLANT DELIVERY TO LEO.
- PAYLOAD POWERED ELECTRIC OTV
  - REQUIRES LEO CONSTRUCTION OF SOLAR ARRAY MODULES
  - LONG TRIP TIME - ECONOMIC PENALTY
  - EXPOSES SOLAR ARRAY TO VAN ALLEN RADIATION
  - LARGE DISTRIBUTED MASS PRESENTS CONTROL PENALTIES

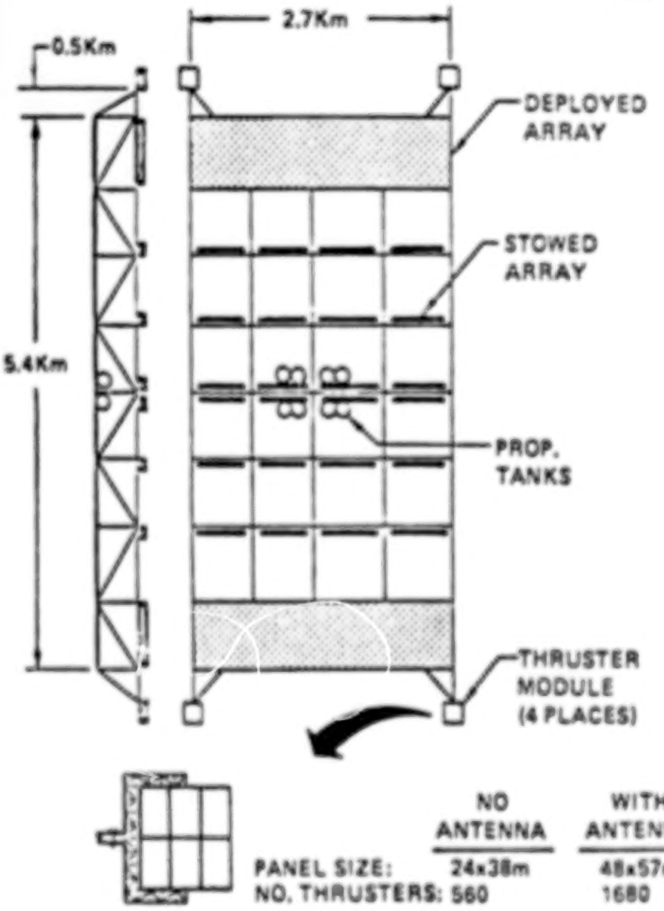
  CONTROL PENALTIES BECAME A DECISION FACTOR.
- INDEPENDENT POWERED ELECTRIC OTV
  - PERMITS GEO CONSTRUCTION
  - LONG TRIP TIME - ECONOMIC PENALTY
  - SPS SOLAR ARRAY PROTECTED FROM RADIATION
  - CONCENTRATED PAYLOAD MASS ALLEVIATES CONTROL PROBLEM

  BETTER MASS DISTRIBUTION DOMINATES PENALTY OF ADDED SOLAR ARRAY.

OTHER OTV POSSIBILITIES:

- SOLAR COLLECTOR, THERMAL CYCLE CONVERSION ELECTRIC PROPULSION
- SOLAR COLLECTOR HEATED HYDROGEN PROPULSION
- LASER HEATED PROPELLANT SYSTEM
- ELECTROMAGNETIC MASS DRIVER

PAYLOAD POWERED OTV


2.7Km
0.5Km
5.4Km
DEPLOYED ARRAY
STOWED ARRAY
PROP. TANKS
THRUSTER MODULE (4 PLACES)

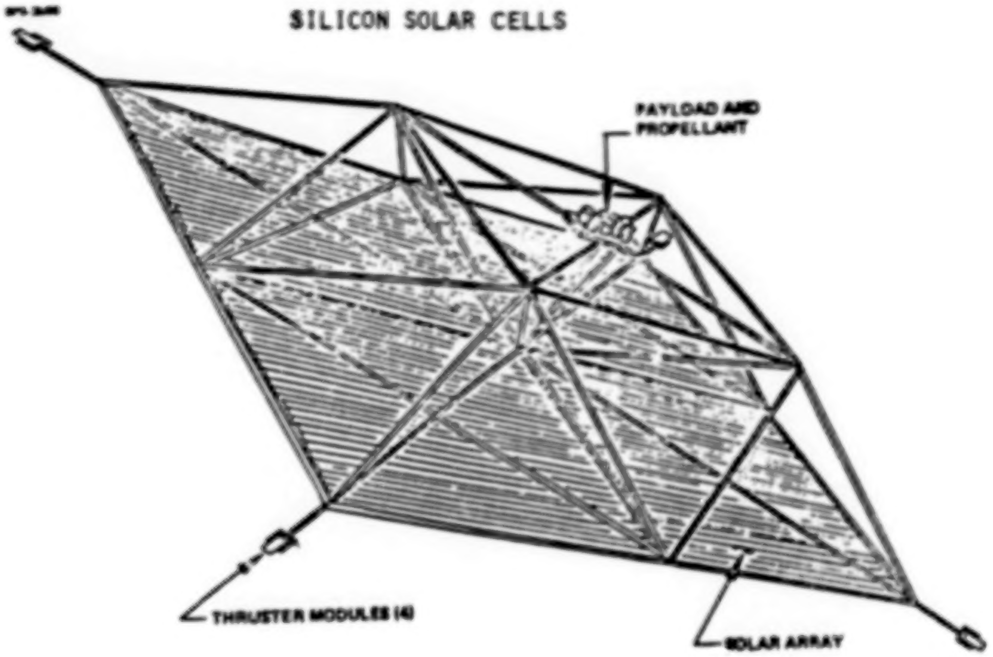
GENERAL CHARACTERISTICS

- 5% OVERSIZING (RADIATION)
- TRIP TIME = 180 DAYS
- ISP = 7000 SEC

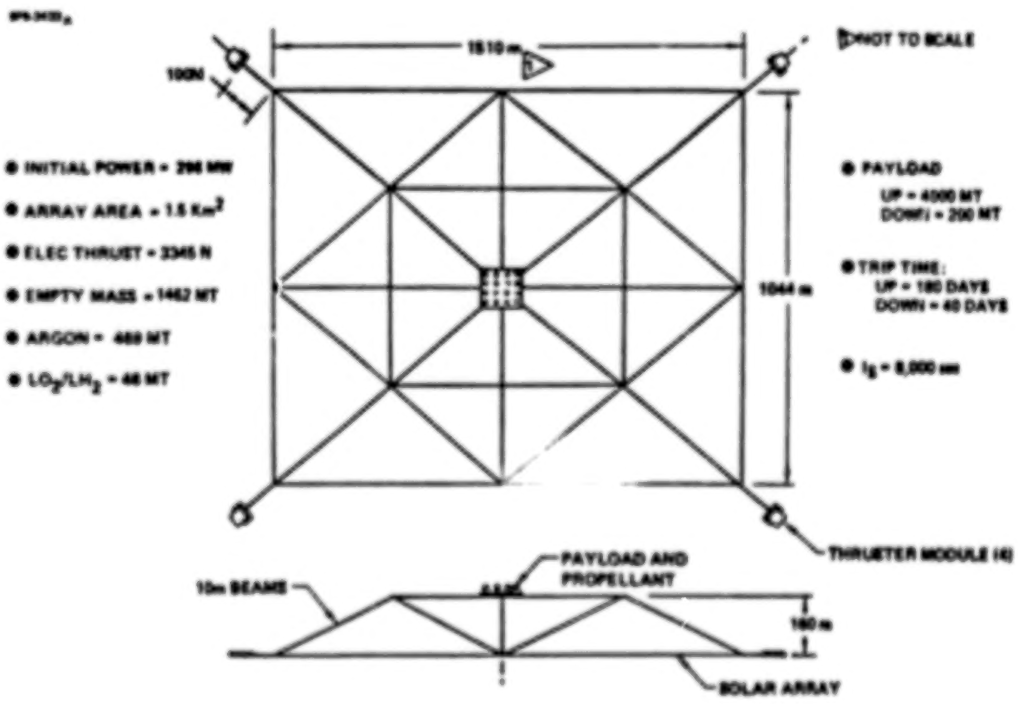
MODULE CHARACTERISTICS	NO ANTENNA	WITH ANTENNA
• NO. MODULES	6	2
• MODULE MASS (10^6KG)	8.7	23.7
• POWER REQ'D (10^6Kw)	0.3	0.81
• ARRAY %	13	36
• OTS DRY (10^6KG)	1.1	2.9
• ARGON (10^6KG)	2.0	5.6
• LO_2/LH_2 (10^6KG)	1.0	2.8
• ELEC THRUST (10^3N)	4.5	12.2
• CHEM THRUST (10^3N)	12.0	5.0

	NO ANTENNA	WITH ANTENNA
PANEL SIZE:	24x38m	48x57m
NO. THRUSTERS:	560	1680

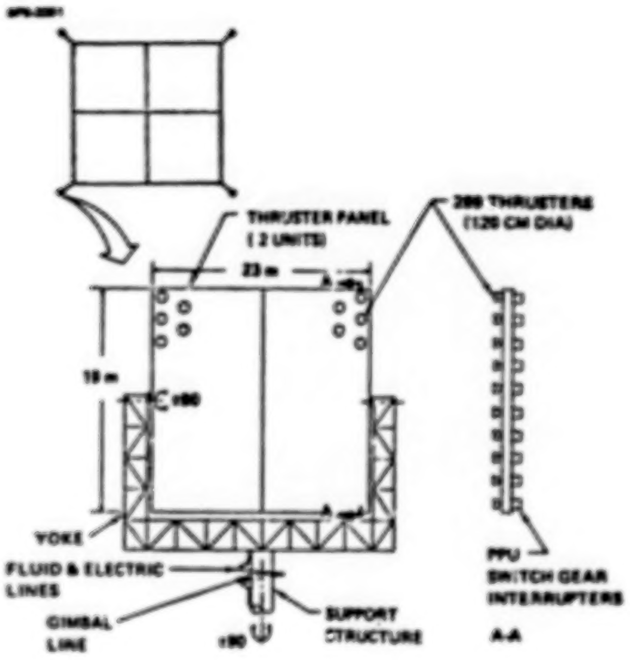
INDEPENDENT POWER EOTV CONFIGURATION CONCEPT - BOEING


SILICON SOLAR CELLS
PAYLOAD AND PROPELLANT
THRUSTER MODULES (4)
SOLAR ARRAY

INDEPENDENT POWER EOTV CONFIGURATION

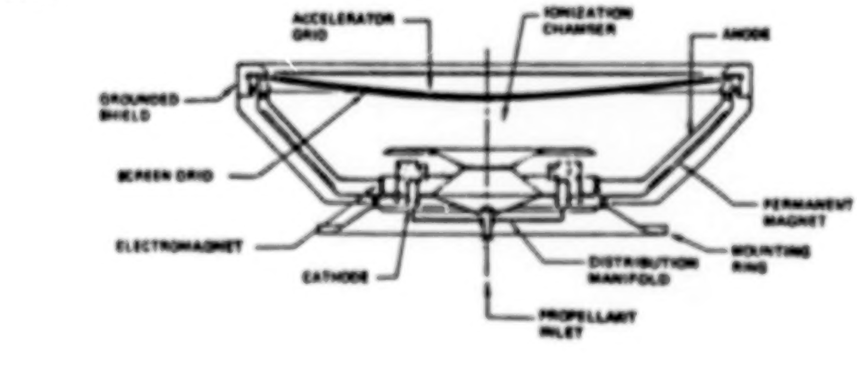

1510 m
NOT TO SCALE
100M
1044 m
● INITIAL POWER = 296 MW
● ARRAY AREA = 1.5 Km²
● ELEC THRUST = 2345 N
● EMPTY MASS = 1462 MT
● ARGON = 469 MT
● LO2/LH2 = 48 MT
● PAYLOAD
UP = 4000 MT
DOWN = 200 MT
● TRIP TIME:
UP = 180 DAYS
DOWN = 40 DAYS
● Isp = 8,000 sec
THRUSTER MODULE (4)
PAYLOAD AND PROPELLANT
10m BEAMS
160 m
SOLAR ARRAY

EOTV PROPULSION SYSTEM


THRUSTER PANEL
(2 UNITS)
288 THRUSTERS
(120 CM DIA)
23 m
19 m
±90
YOKE
FLUID & ELECTRIC
LINES
GIMBAL
LINE
±90
SUPPORT
STRUCTURE
PPU
SWITCH GEAR
INTERRUPTERS
A-A

- THRUSTER POWER SUPPLY
  - DIRECTLY FROM ARRAY
    - NO PROCESSING
    - NO REGULATION
  - NO PROCESSING
    - ARRAY REGULATION
  - ✓ PROCESS ALL POWER
- TYPE OF PROCESSING
  - MOTOR/GENERATOR
  - ✓ SOLID STATE
- PROCESSING THERMAL CONTROL
  - ACTIVE RADIATOR
  - 918 m^2
  - LIMIT ELECTRONICS TO 200°C

120 cm ION THRUSTER


ACCELERATOR
GRID
IONIZATION
CHAMBER
ANODE
GROUNDED
SHIELD
SCREEN GRID
PERMANENT
MAGNET
ELECTROMAGNET
MOUNTING
RING
CATHODE
DISTRIBUTION
MANIFOLD
PROPELLANT
INLET

SELECTED 120 CM ARGON ION THRUSTER CHARACTERISTICS

FIXED CHARACTERISTICS	
BEAM CURRENT:	80.0 AMPS.
ACCEL. VOLTAGE:	500.0 V.
DISCHARGE VOLTAGE:	30.0 V. (FLOATING)
COUPLING VOLTAGE:	11.0 V.
DBL. ION RATES:	0.16 (J2/J1)
NEUTRAL EFFLUX:	4.8384 AMP. EQUIV.
DIVERGENCE:	0.98
DISCHARGE LOSS:	187.3 EV/ION
OTHER LOSS:	1758.0 W.
UTILIZATION:	0.882 W.
LIFE:	8000 HR.
*WEIGHT:	50. KG.
SELECTED CHARACTERISTICS	
SCREEN (BEAM) VOLTAGE:	1700 V.
INPUT POWER:	130 KW
THRUST:	2.9 N
EFFICIENCY:	78

*WEIGHT PREDICTION COURTESY OF T. MASEK OF HRL.

EOTV MASS STATEMENT

POWER GENERATION AND DISTRIBUTION		951,000	
SOLAR ARRAY-SILICON	780,000		
STRUCTURE	122,000		
POWER DISTRIBUTION	42,000		
ENERGY STORAGE	7,000		
ELECTRIC PROPULSION		496,000	
THRUSTERS	79,000		
POWER CONDITIONING	219,000		
THERMAL CONTROL	88,000		
STRUCTURES & MECHANISM	61,000		
PROPELLANT FEED	49,000		
AUXILIARY SYSTEMS		15,000	
DRY WEIGHT			1,462,000
PROPELLANT			515,000
ARGON	469,000		
HYDROGEN	6,600		
OXYGEN	39,400		
PAYLOAD (GROSS)			4,000,000
TOTAL START BURN MASS.			5,977,000

EOTV AVERAGE COST PER UNIT

FLIGHT UNIT			247.0 M
• POWER GENERATION & DISTRIBUTION		99.7	
SOLAR ARRAY	79.6		
STRUCTURE	12.2		
POWER DISTRIBUTION	1.6		
ENERGY STORAGE	6.4		
• ELECTRIC PROPULSION		141.0	
THRUSTERS	15.4		
POWER CONDITIONING	87.2		
THERMAL CONTROL	22.1		
STRUCTURES & MECHANISMS	11.3		
PROPELLANT SYSTEM	5.0		
• AVIONICS		6.5	
PROGRAMMATICS			36.6
TOTAL			$283.6 M

EOTV - PER FLIGHT COST

HARDWARE

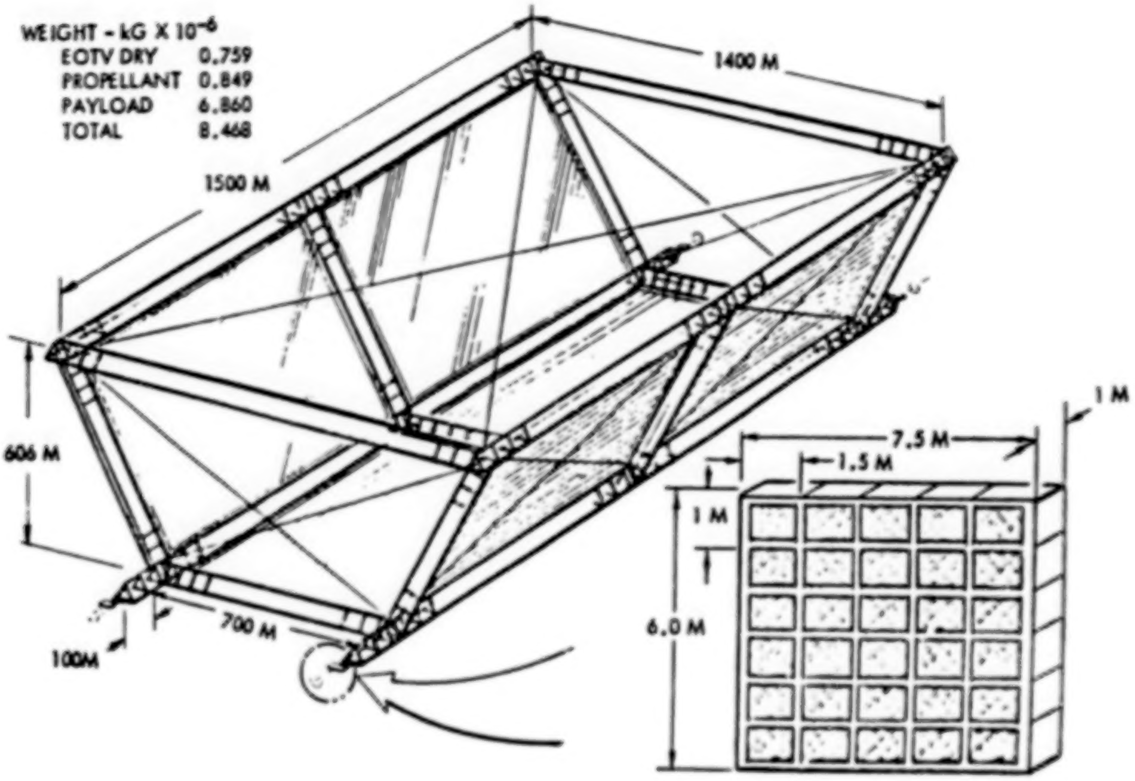
$$\frac{\text{AVERAGE VEHICLE COST}}{\text{FLIGHTS PER VEHICLE}} = \frac{283,600}{10} = \$28,400\text{K}$$

PROPELLANT

ARGON	470 MT @ $1/KG	470
O_2	39 MT @ $.037/KG	1
H_2	7 MT @ $1.53/KG	11

REFURBISHMENT	11,300
PROGRAM SUPPORT	500
TOTAL COST PER FLIGHT.	$40,682K

INDEPENDENT POWER EOTV CONCEPT - ROCKWELL
(GALLIUM ARSENIDE)


WEIGHT - kG X 10^{-6}
EOTV DRY 0.759
PROPELLANT 0.849
PAYLOAD 6.860
TOTAL 8.468
1500 M
1400 M
606 M
700 M
100M
1 M
7.5 M
1.5 M
1 M
6.0 M

ARGON ION THRUSTER CHARACTERISTICS - ROCKWELL

MAXIMUM TOTAL VOLTAGE, VOLT	4405
MAXIMUM OPERATING TEMP, °K	1330
SCREEN GRID VOLTAGE, VOLT	1880
ACCELERATOR GRID VOLTAGE, VOLT	-2525
BEAM CURRENT, AMP	1500
BEAM POWER, WATT	2.8×10^6
SPECIFIC IMPULSE, SEC	7963
THRUST, NEWTON	56.26

BEAM CURRENT VS. THRUSTER LIFE ASSUMPTIONS

BEAM CURRENT -	80 AMPS/M^2	1500 AMPS/M^2
THRUSTER LIFE -	8000 HOURS	4000 HOURS

EOTV KEY ISSUE TECHNOLOGY NEEDS - GENERAL:

- CAPABILITY FOR COMPREHENSIVE ANALYSES OF COMPLEX, EXTREMELY LARGE STRUCTURES UNDER GRAVITY GRADIENT LOADS, NON-CONSTANT APPLIED FORCES, AND THERMAL TRANSIENTS. (STRUCTURAL CANNOT BE TESTED UNTIL IT IS CONSTRUCTED IN SPACE.)
- SELECTION OF STRUCTURAL MATERIALS FOR THER THERMAL, VACUUM, AND RADIATION ENVIRONMENT OF LEO-GEO FLIGHT. MEASUREMENT OF REQUIRED PROPERTIES FOR DESIGN.
- ANNEALING OF RADIATION DAMAGE IN SOLAR CELLS.
- HIGHLY RELIABLE, REDUNDANT ATTITUDE-CONTROL SYSTEM WHICH GUARANTEES STABILITY DURING OCCULATION OF THE SUN.
- AUTONOMOUS NAVIGATION, GUIDANCE, AND CONTROL SYSTEM.
- MEANS TO ASSURE AGAINST RE-ENTRY FROM LOW-EARTH ORBIT.

ELECTRIC PROPULSION - KEY ISSUES

- TECHNOLOGY FOR SCALING ION THRUSTERS FROM 30 cm TO 100 cm AND ABOVE.
    - GRID STABILITY
    - MULTIPLE CATHODE DESIGN
- REPLACEMENT OF MERCURY BY ARGON AS PROPELLANT.
- IONOSPHERIC EFFECTS OF ARGON.
- SYSTEMS DESIGN TRADE DATA TO SELECT:
    - THRUSTER LIFE
    - POWER
    - THRUST
    - Isp

LOW-THRUST CHEMICAL ROCKET ENGINE STUDY

Joseph A. Mellish
Aerojet Liquid Rocket Company

Low-Thrust Engine Study Program, Contract NAS 3-21940

A number of studies have forecast the need for large space structures such as microwave antennas and reflectors in geosynchronous equatorial orbit (GEO). These structures would be launched to low earth orbit (LEO) in a stowed condition using the Space Shuttle and subsequently transferred to GEO using a high energy space propulsion system. There are two options available for placement of these types of payloads in GEO. In the first option, the LEO-to-GEO transfer would be accomplished with the payload in the stowed condition, followed by manned or automated deployment and assembly in GEO. Either high or low thrust could be used for the transfer. In the second option, manned or automated deployment and assembly would be carried out in LEO, followed by a LEO-to-GEO transfer with the payload in the assembled condition. Here, low thrust would be required in order to preclude high inertia loading which would cause damage to the assembled payload. Chemical engine systems suitable for the low-thrust option have not received in-depth attention and it is the purpose of this work to provide the data necessary for orbit-transfer-vehicle studies utilizing low-thrust chemical propulsion.

The major objectives of this Low-Thrust Chemical Rocket Engine Study are to provide parametric data and preliminary designs on liquid rocket engines for low thrust cargo orbit-transfer-vehicles and to identify those items where technology is required to enhance the designs. These data and the systems analyses will ultimately lead to the identification of low-thrust OTV engine requirements so that the engine design and development phase can be initiated.

Low-Thrust Engine Study Program
Contract NAS 3-21940

- APPLICATION

  CARGO ORBIT-TRANSFER VEHICLE (COTV)

- PRIMARY OBJECTIVES

  1. PROVIDE PARAMETRIC DATA AND PRELIMINARY DESIGNS ON LIQUID ROCKET ENGINES
  2. IDENTIFY TECHNOLOGY REQUIREMENTS

Specific Objectives

Specific study objectives are:

- Provide fundamental propellant property, combustion property and performance data for O_2/H_2, O_2/RP-1 and O_2/CH_4 engine concepts.
- Establish the combined thrust level and chamber pressure range over which film and regeneratively cooled low-thrust chamber designs are feasible.
- Devise six engine system concepts. (Initial efforts considered only conventional cooling schemes and will be updated on about 15 July 1980 to include advanced cooling concepts).
- Generate parametric performance, weight and envelope data for viable concepts based upon historical data and conceptual evaluations. The first data dump (9 May 1980) was based upon conventional cooling techniques and will be updated to include the advanced cooling schemes.
- Select concepts and design points for preliminary design.
- Prepare preliminary designs of two concepts. One uses oxygen hydrogen propellants and the other oxygen/hydrocarbon.
- Update the parametric data based upon the preliminary design results and provide this data in a format suitable for use by OTV vehicle system contractors.

Specific Objectives

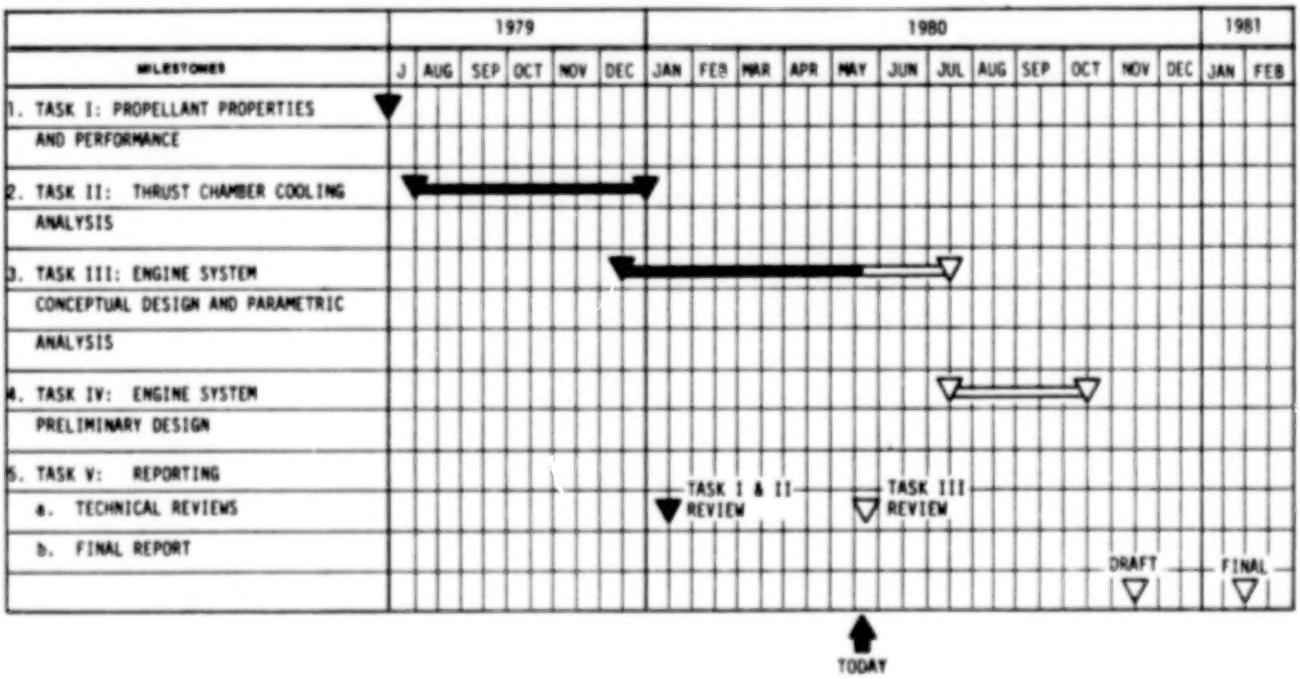
	DELIVERY DATES
1. PROVIDE PROPELLANT PROPERTY AND PERFORMANCE DATA (TASK I)	10 SEPT 1979
2. ESTABLISH FEASIBLE THRUST AND CHAMBER PRESSURE RANGES FOR FILM AND REGEN COOLING (TASK II)	17 JAN 1980
3. EVALUATE SYSTEM CONCEPTS AND DEVISE ENGINE CONCEPTS (TASK III)	23 MAY 1980
4. PROVIDE PARAMETRIC DATA (TASK III)	9 MAY & 15 JULY 1980
5. PREPARE PRELIMINARY DESIGNS OF TWO CONCEPTS (TASK IV)	15 OCT 1980
6. UPDATE PARAMETRIC DATA (TASK IV)	15 OCT 1980

Low-Thrust Chemical Rocket Engine Study Schedule

The current study schedule is shown on the figure. This schedule reflects the changes to accommodate the additional Task III work involved in evaluating advanced cooling schemes.

In Task I, properties and/or theoretical performance of the subject propellants and propellant combinations over the low-thrust range of interest were determined. Task II involved analyses to establish the combined thrust level and chamber pressure range over which film and regeneratively cooled low-thrust chamber designs are feasible. In Task III, engine system concepts are devised and evaluated over the thrust chamber cooling feasibility range to establish a feasible design range for the engine system concepts. Parametric data (performance, weight, etc.) are generated for the viable concepts to assist in the selection of concepts and design points for preliminary design. In Task IV, preliminary design will be accomplished on two concepts (one hydrogen-oxygen and one hydrocarbon-oxygen) and the parametric data for the selected concepts will be updated to reflect the results of the preliminary design effort.

Low-Thrust Chemical Rocket Engine Study Schedule



Low-Thrust Engine Study Cases

The original study guidelines specified that the engines would be either regeneratively cooled or film cooled and combined regen/film cases were not included in the analyses. In addition, only the fuels were considered as coolants. The contract is currently being modified to include other cooling schemes.

Low-Thrust Engine Study Cases

Propellant Combination	O/F	Cooling Method	Coolant	Thrust Study Range (LBF)	Chamber Pressure Study Range (PSIA)
H_2/O_2	6.0	Regen	H_2	100-3000	20-1000
H_2/O_2	6.0	Film	H_2	100-3000	20-1000
RP-1/O_2	3.0	Regen	RP-1	100-3000	20-1000
RP-1/O_2	3.0	Film	RP-1	100-3000	20-1000
CH_4/O_2	3.7	Regen	CH_4	100-3000	20-1000
CH_4/O_2	3.7	Film	CH_4	100-3000	20-1000

Task I Propellants and Parametric Ranges

The thrust ranges were shown on the previous chart and other ranges are shown here. One dimensional equilibrium (ODE) specific impulse data was generated over a range of area ratios from 1 to 1000 although a nominal value of 400:1 is used in the conduct of Tasks II and III. Mixture ratio (O/F) ranges are also shown and the nominal values for each propellant combination were shown on the previous chart.

Task I Propellants And Parametric Ranges

- PROPELLANTS - O_2, H_2, RP-1, CH_4
- PROPELLANT COMBINATIONS

  O_2/H_2, O_2/RP-1, O_2/CH_4
- PARAMETRIC RANGES

  CHAMBER PRESSURE: 20 TO 1000 PSIA

  AREA RATIO: 1 TO 1000

  MIXTURE RATIO

    O_2/H_2: 4 TO 7

    O_2/RP-1: 2.6 TO 3.2

    O_2/CH_4: 3.4 TO 4.0

Task II Thrust Chamber Cooling Analysis Objectives

The key objective of the Task II cooling analysis was to identify feasible operating ranges using conventional cooling techniques and design criteria. As mentioned previously, the study is being extended to include advanced cooling methods.

Task II Thrust Chamber Cooling Analysis Objectives

- DETERMINE THE COMBINED THRUST LEVEL AND CHAMBER PRESSURE RANGE OVER WHICH LOW-THRUST CHAMBER DESIGNS ARE FEASIBLE USING CONVENTIONAL COOLING METHODS AND DESIGN CRITERIA.
- PROVIDE HEAT TRANSFER AND HYDRAULIC PARAMETRIC DATA FOR USE IN ENGINE SYSTEM ANALYSIS EFFORT IN TASKS III AND IV.

Task II Cooling Analysis Guidelines

Some of the cooling analysis guidelines specified by the SOW are shown. The chambers analyzed are a slotted design configuration and the study also imposed practical limits on these designs such as,

Minimum slot width = .03 in.
Maximum slot depth/width = 4 to 1
Minimum web thickness = .03 in.
Minimum wall thickness = .025 in.
Minimum channel depth = .035 in.

Task II Cooling Analysis Guidelines

- 90% BELL NOZZLES ($\epsilon = 400:1$)
- COOLANT INLET TEMPERATURE

  H_2 = 37.8 °R

  RP-1 = 537 °R

  CH_4 = 201 °R

- POSSIBLE BENEFIT OF CARBON DEPOSITION ON HOT GAS SIDE WALL SHALL BE NEGLECTED.
- COOKING LIMIT

  RP-1 = 1010°R

  CH_4 = 1760°R

- SERVICE LIFE

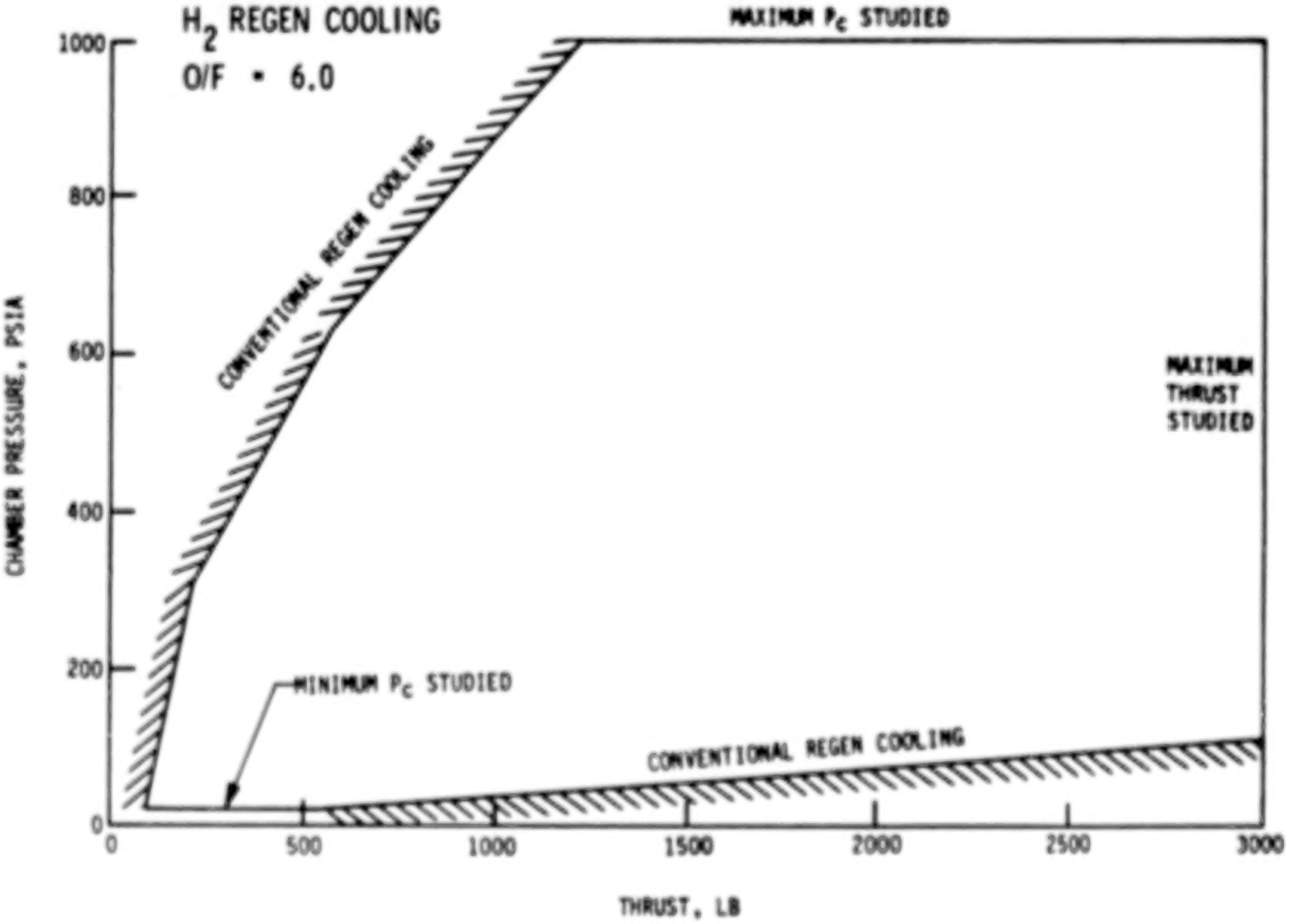
  FIVE THERMAL CYCLES TIMES A SAFETY FACTOR OF FOUR.

  $$\text{ENGINE RUN TIME} = \frac{\text{5000 LB-HRS}}{\text{THRUST, LBS}}$$

O_2/H_2 Regen Cooled Engine Operating Region

Hydrogen provided the largest operating map on a thrust-chamber pressure plot. Channel Mach number limits and channel depth considerations constrain operation at low thrust-high Pc and high thrust-low Pc combinations. The feasible cooling map with hydrogen covers both the supercritical and subcritical pressure regimes. The critical pressure of hydrogen is 188 psia and coolant jacket exit pressure was held above this value to obtain a practical design solution. This will penalize pressure-fed systems with regen cooled engines because of the high hydrogen tank pressure. Thrusts greater than 3000 lb and chamber pressures greater than 1000 psia were not considered in this study although they would be feasible.

O_2/H_2 Regen Cooled Engine Operating Region



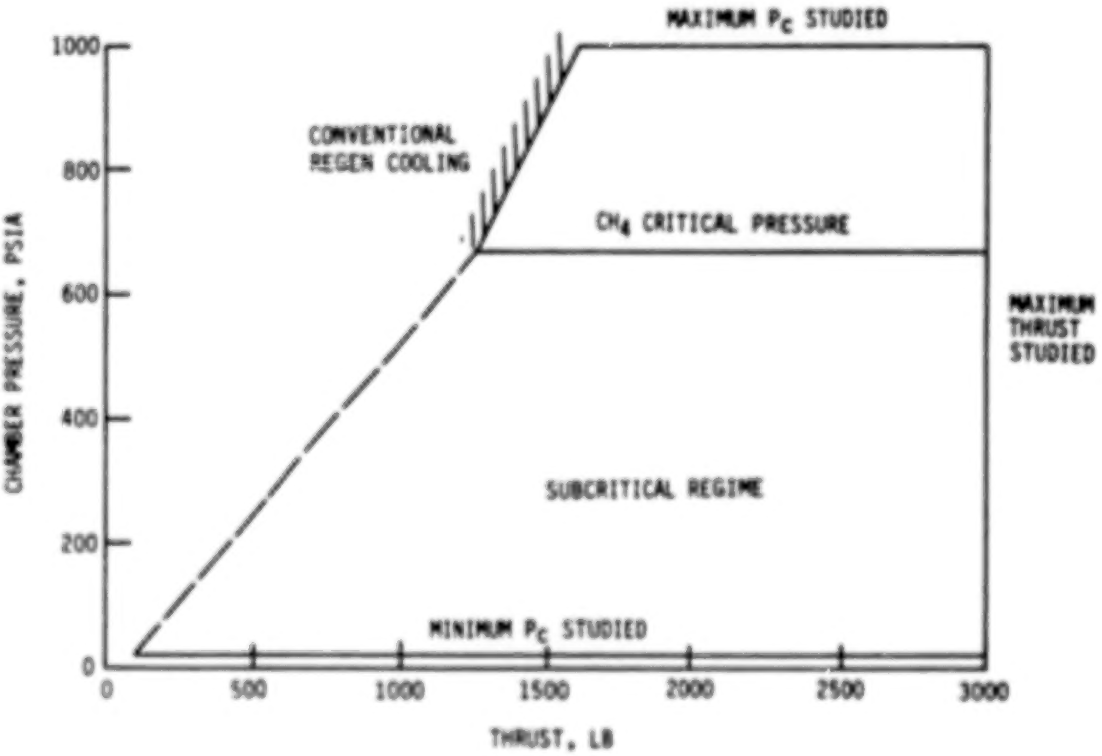
O_2/CH_4 Regen Cooled Engine Operating Region

Methane provided a smaller feasible cooling map with the channel Mach number limiting operation to a higher thrust and pressure region. Feasible designs could not be obtained with the CH_4 below its critical pressure (i.e. 667 psia). However, engine system analysis in the sub-critical pressure regime was continued by keeping the coolant jacket outlet pressure above critical. This places the burden upon the CH_4 pumping system.

O_2/CH_4 Regen Cooled Engine Operating Region

CH_4 REGEN COOLING

O/F = 3.7



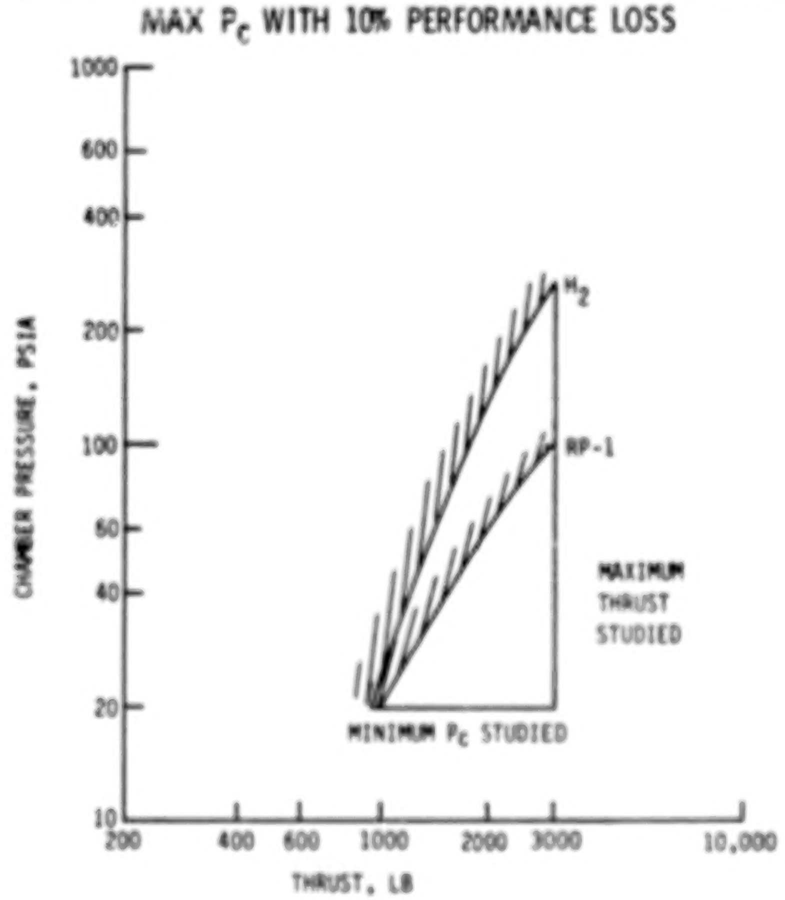
Film Cooling Analyses Results

The results of the film cooling studies to establish the upper chamber pressure limit, based upon a 10% performance degradation, are shown. This performance degradation is based upon a comparison to the performance of an engine requiring no film cooling.

Hydrogen and RP-1 cannot be used as film coolants at thrusts below about 1000 lbf and their chamber pressure ranges are very limited. Hydrogen is penalized by the low wall temperature (1800°F) obtainable with compatible materials, and RP-1 is penalized by the long chamber lengths required to achieve a minimum study specified energy release efficiency of 98%. RP-1 film cooled engines were dropped from further study because of this small operating range. Lower limit chamber pressures corresponding to a 3 percent performance degradation were found to be approximately at or below the specified minimum chamber pressure of 20 psia.

The feasibility of methane film cooling is highly dependent upon the kinetics of the methane decomposition. However, this analysis was beyond the scope of the current effort. The sensitivity of the results to the chemistry model assumption was assessed at a thrust of 1000 lbf and a chamber pressure of 300 psia. Assuming no CH_4 decomposition and thus, no coolant reaction with the entrained core gases, the coolant requirement exceeds 50% of the fuel and the performance loss exceeds 20%. With the complete decomposition assumption, the required coolant flow is about 33% of the fuel flow and performance loss is 10%. Because of this uncertainty, NASA/LeRC has elected to temporarily drop CH_4 film cooled engines from the analysis. Data is required to verify the models.

Film Cooling Analyses Results



Cooling Analyses Conclusions

The study showed that regen cooling with RP-1 was not feasible over the entire thrust and chamber pressure ranges. The thermal data showed that the RP-1 bulk temperature exceeded the study coking temperature limit of 1010°R. This result might change if chamber coatings, possible benefits from carbon deposition on the hot gas side wall or a purified RP-1 were considered. These were not within the current study scope but will be considered in the extension efforts.

Based upon the results presented, O_2/H_2 and O_2/CH_4 regen engine systems and O_2/H_2 film cooled engines were selected for further study in the system analysis. Advanced cooling schemes and investigations will also be considered for all propellants in further study efforts.

Cooling Analyses Conclusions

- VIABLE CONCEPTS WITH CONVENTIONAL COOLING METHODS:
  - O_2/H_2, H_2 REGEN COOLED
  - O_2/H_2, H_2 FILM COOLED
  - O_2/CH_4, CH_4 REGEN COOLED
- RP-1 REGEN COOLANT TEMPERATURE EXCEEDED 1010°R COKING LIMIT OVER ENTIRE THRUST AND PRESSURE RANGES.
- EXTEND STUDY DESIGN CRITERIA TO CONSIDER ADVANCED COOLING SCHEMES.

Task III Engine System Conceptual Design and Parametric Analysis

Task III involves the screening and evaluation of candidate concepts, the selection of concepts for further analyses, generation of parametric data for the concepts and the selection of two concepts for preliminary design analyses in Task IV. The concepts undergoing evaluation are presented on the following six charts. The thrust and chamber pressure operating ranges are as defined by the cooling analysis results, unless this range is further restricted by cycle or concept limits which are determined in conducting this task. Parametric data is generated over the feasible thrust and chamber pressure ranges at the nominal propellant combination mixture ratios and an area ratio of 400:1. The concept weights are estimated by scaling historical component weight data in this size range.

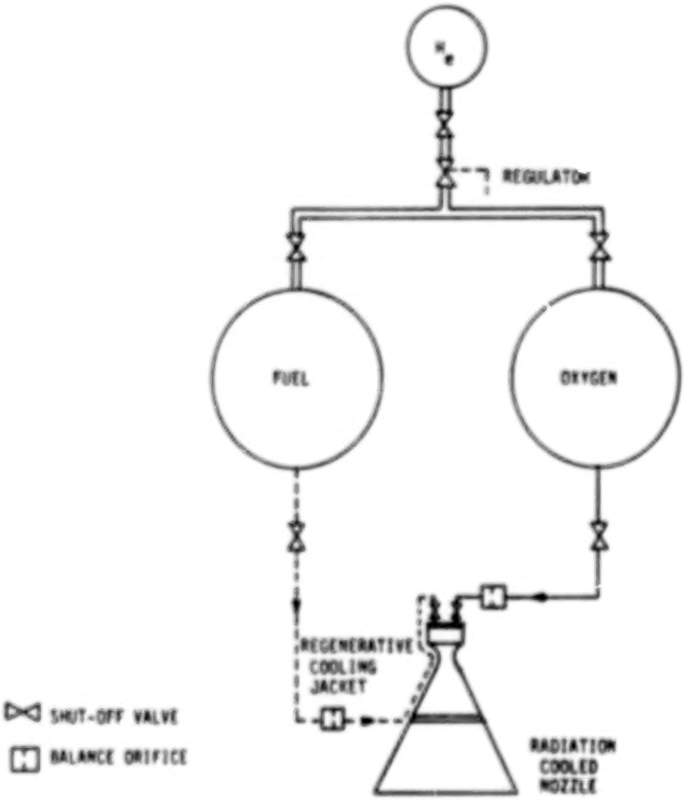
Task III Engine System Conceptual Design And Parametric Analyses Objectives

- ASSESS THE FEASIBILITY OF VARIOUS DESIGN APPROACHES.
- ESTABLISH OPERATING RANGES.
- DETERMINE ADVANTAGES AND DISADVANTAGES OF CONCEPTS.
- ASSESS TECHNOLOGY REQUIREMENTS.
- PREPARE PARAMETRIC DATA (ϵ = 400) (WEIGHT, PERFORMANCE & ENVELOPE).

Pressure-Fed Concept

The simple pressure-fed system concept is shown on the figure. In this concept the engine run tanks are pressurized to the required pressure levels by a regulated helium source. It should be noted that the concept is applicable to both regen and film cooled engines.

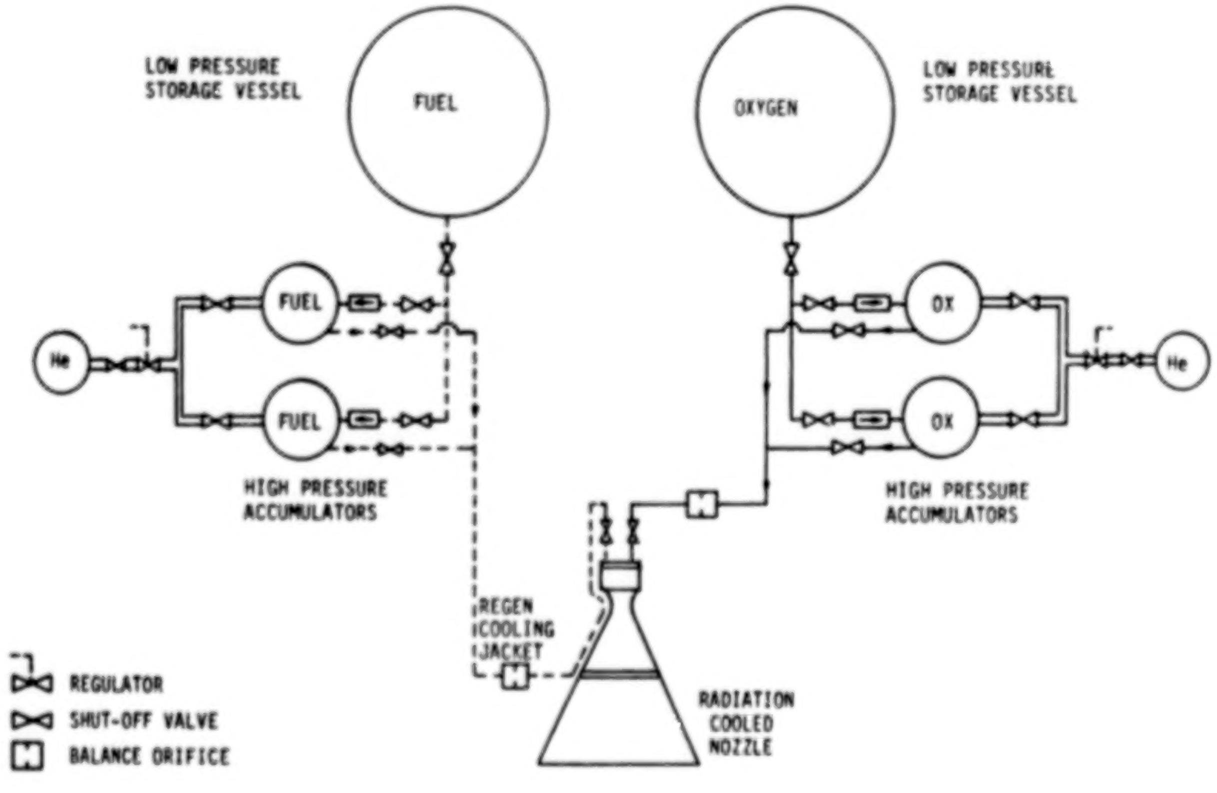
Pressure-Fed Concept


He
REGULATOR
FUEL
OXYGEN
REGENERATIVE
COOLING
JACKET
SHUT-OFF VALVE
BALANCE ORIFICE
RADIATION
COOLED
NOZZLE

Parallel Accumulator Concept

A parallel pressurized tank concept is shown on the schematic. In this concept, both the fuel and oxygen are stored in low pressure main propellant tanks. Two small parallel accumulators in each propellant feed system are located downstream of these main propellant tanks. These accumulators are alternately filled from the main propellant tank and pressurized to provide the engine propellant supply. When the propellant is expelled, the tank is vented and then refilled from the main tank. While one tank is being filled, the engine runs off of the parallel tank. The advantage of this system over the basic pressure-fed concept is a reduction in the high pressure tankage weight. The accumulators are sized to provide the apogee burn. Again, the engine can be either regen or film cooled.

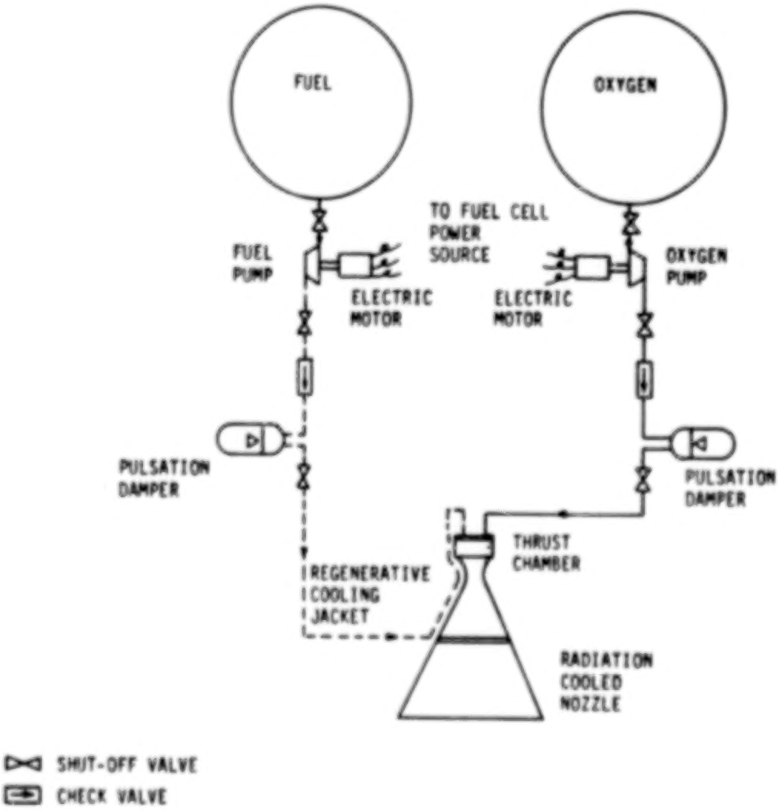
Parallel Accumulator Concept (Pressure-Fed)


LOW PRESSURE STORAGE VESSEL
FUEL
OXYGEN
LOW PRESSURE STORAGE VESSEL
He
FUEL
FUEL
OX
OX
He
HIGH PRESSURE ACCUMULATORS
HIGH PRESSURE ACCUMULATORS
REGEN COOLING JACKET
RADIATION COOLED NOZZLE
REGULATOR
SHUT-OFF VALVE
BALANCE ORIFICE

Auxiliary Power Source (Fuel Cells) Concept

The figure shows a pump-fed concept in which the pumps are driven by electric motors with fuel cells as the power source. Analysis has indicated that the weight of batteries is prohibitive. The concept shown has a pulsation damper (very small accumulator) downstream of the pumps. This component will be required if positive displacement pumps are selected in component screening analysis. This concept is also applicable with film or regen cooling.

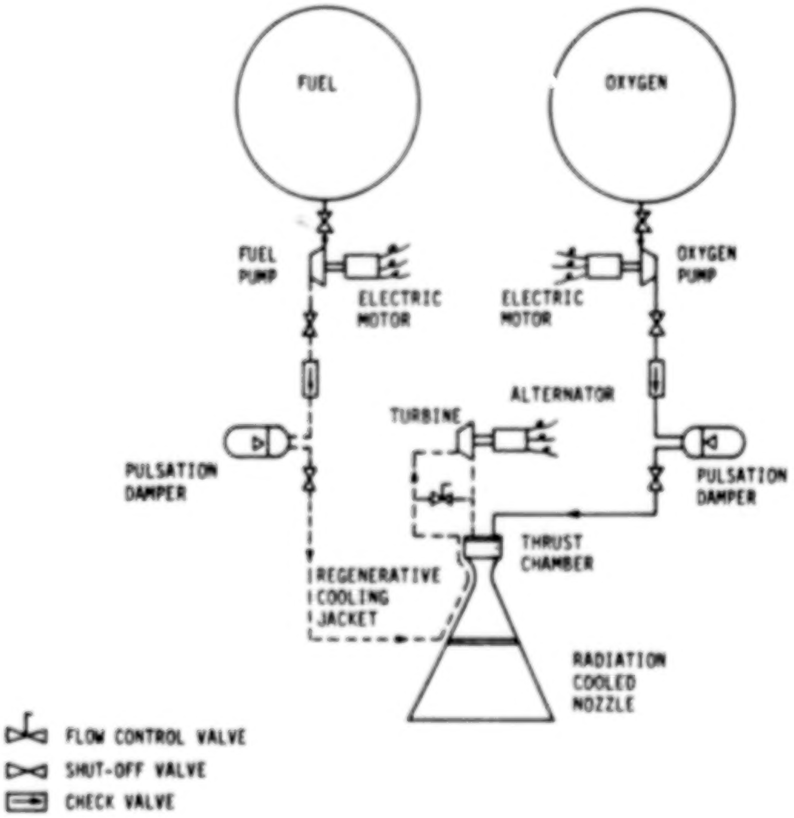
Auxiliary Power Source (Fuel Cells) Concept


FUEL
OXYGEN
TO FUEL CELL POWER SOURCE
FUEL PUMP
ELECTRIC MOTOR
ELECTRIC MOTOR
OXYGEN PUMP
PULSATION DAMPER
PULSATION DAMPER
THRUST CHAMBER
REGENERATIVE COOLING JACKET
RADIATION COOLED NOZZLE
SHUT-OFF VALVE
CHECK VALVE

Turboalternator Concept

The figure shows a pump-fed concept with an electric motor drive using a turboalternator as the power source. This concept has potential application with heated hydrogen or methane as the turbine drive fluid. A small amount of the heated fuel bypasses the turbine. This bypass flow provides the power control. Cycle power balances were performed to determine if the maximum operating chamber pressure of this system is different than the cooling limits. This is discussed with a later chart.

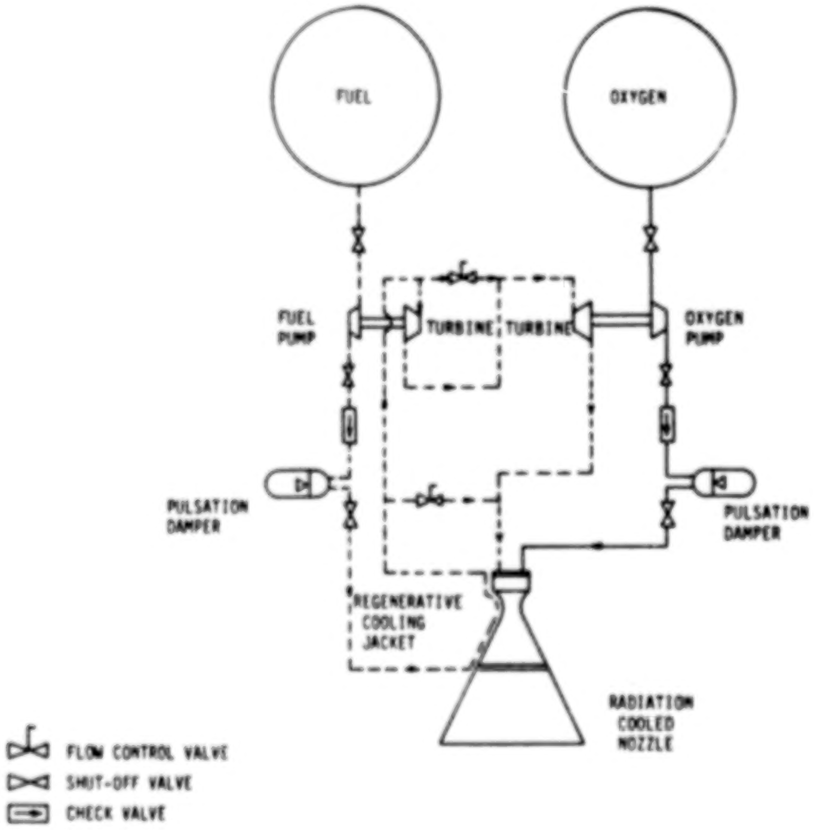
Turboalternator Concept


FUEL
OXYGEN
FUEL PUMP
ELECTRIC MOTOR
ELECTRIC MOTOR
OXYGEN PUMP
ALTERNATOR
TURBINE
PULSATION DAMPER
PULSATION DAMPER
THRUST CHAMBER
REGENERATIVE COOLING JACKET
RADIATION COOLED NOZZLE
FLOW CONTROL VALVE
SHUT-OFF VALVE
CHECK VALVE

Expander Cycle Concept

An expander cycle pump-fed concept is shown on the schematic. This concept is also applicable with heated hydrogen or methane as the drive fluid for the turbines. A series turbines cycle arrangement was selected because the full flow oxygen turbine is much more efficient than the extremely low flow oxygen turbine in a parallel arrangement. The fuel turbine bypass valve shown on the figure is used to provide mixture ratio control and the valve bypassing flow around both turbines is for power control. This is the simpliest pump-fed system because it does not require any additional components.

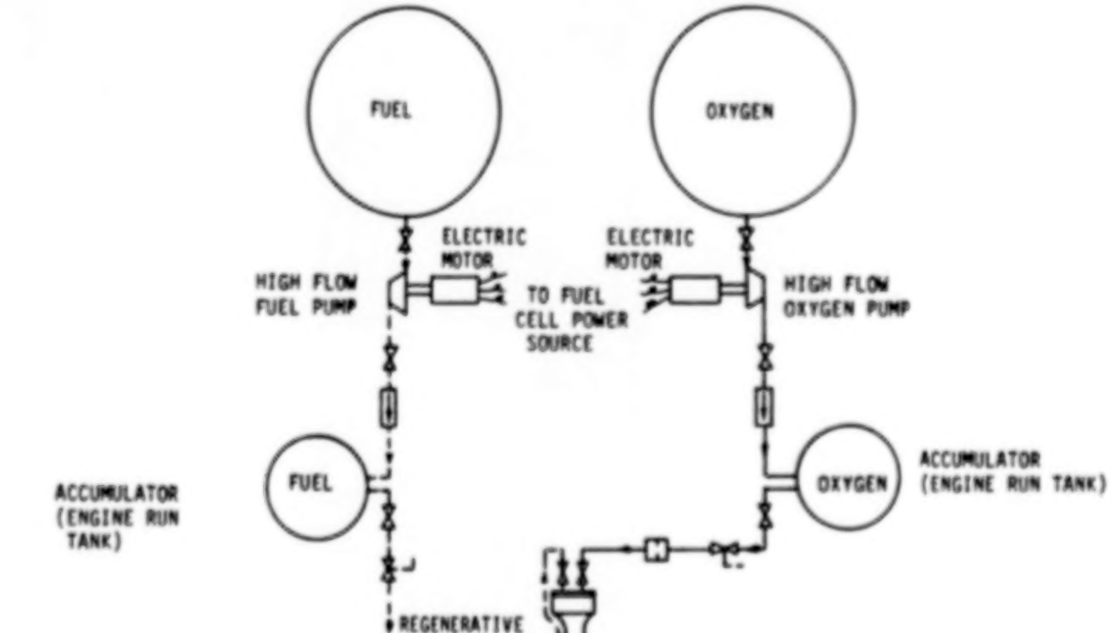
Expander Cycle Concept


FUEL
OXYGEN
FUEL PUMP
TURBINE
TURBINE
OXYGEN PUMP
PULSATION DAMPER
PULSATION DAMPER
REGENERATIVE COOLING JACKET
RADIATION COOLED NOZZLE
FLOW CONTROL VALVE
SHUT-OFF VALVE
CHECK VALVE

Pump-Filled Feed System Tank Concept

A pump-filled tanks engine feed system concept is shown schematically. In this concept, the engine run tanks are filled by pumps from the low pressure main vessels during mission coast periods. The possible advantage of this concept is that the pump flows can be much higher than the engine flows which may provide a more suitable operating regime for the pumps (i.e., the pump design is not restricted by the engine thrust level). A regulator is shown downstream of the engine run tanks to maintain constant engine pressures. Without this regulator, the chamber pressure and engine thrust would decay as the propellant is expelled. This system is applicable with regen or film cooled engines.

Pump-Filled Feed System Tank Concept


FUEL
OXYGEN
ELECTRIC MOTOR
ELECTRIC MOTOR
HIGH FLOW FUEL PUMP
TO FUEL CELL POWER SOURCE
HIGH FLOW OXYGEN PUMP
FUEL
ACCUMULATOR (ENGINE RUN TANK)
OXYGEN
ACCUMULATOR (ENGINE RUN TANK)
REGENERATIVE COOLING JACKET
RADIATION COOLED NOZZLE
REGULATOR
SHUT-OFF VALVE
CHECK VALVE
BALANCE ORIFICE

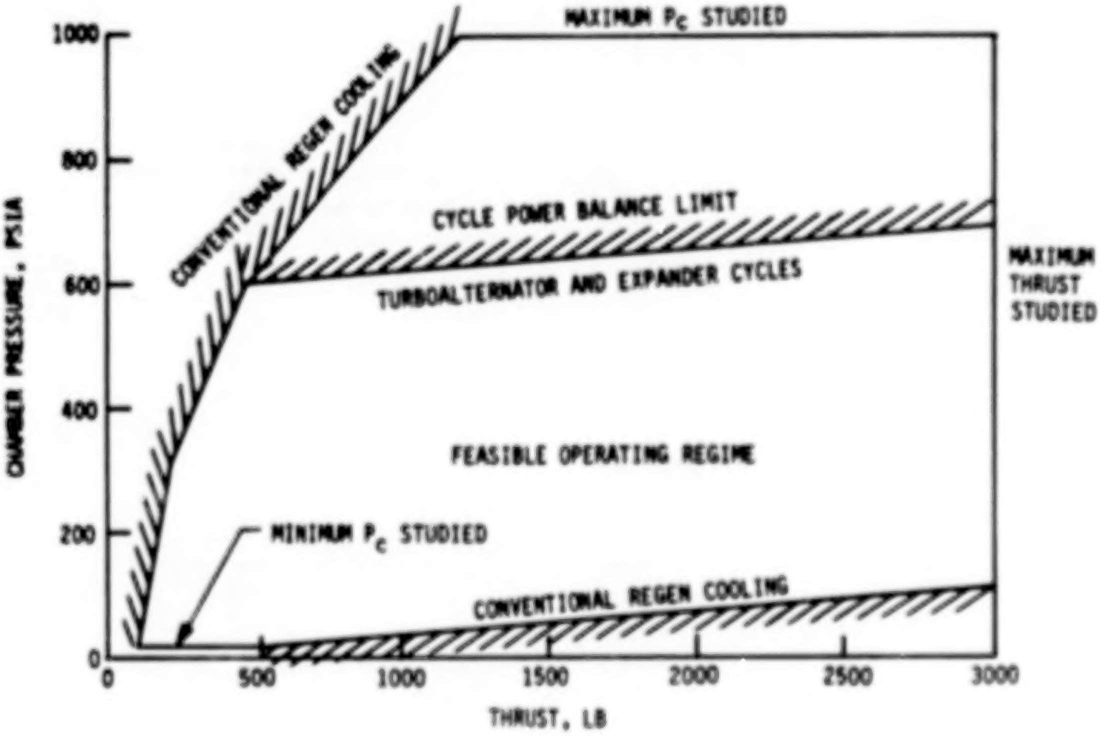
O_2/H_2 Turboalternator and Expander Cycle Operating Regime

The figure shows cycle power balance limits for both the turboalternator and expander cycle concepts superimposed on the feasible cooling map. For these two cycles, the operating region is reduced even further. The power balance is limited by the coolant jacket pressure drop, turbine inlet temperature and component efficiencies. For pump-fed systems using an auxiliary power source (i.e. fuel cells), operation to 1000 psia is possible although the power requirements are very large. Engine parametric data was run over the feasible operating regimes as defined by either the cooling or power balance limits. Advanced cooling schemes may extend these limits.

O_2/H_2 Turboalternator And Expander Cycle Operating Regime

H_2 REGEN COOLED

O/F • 6.0



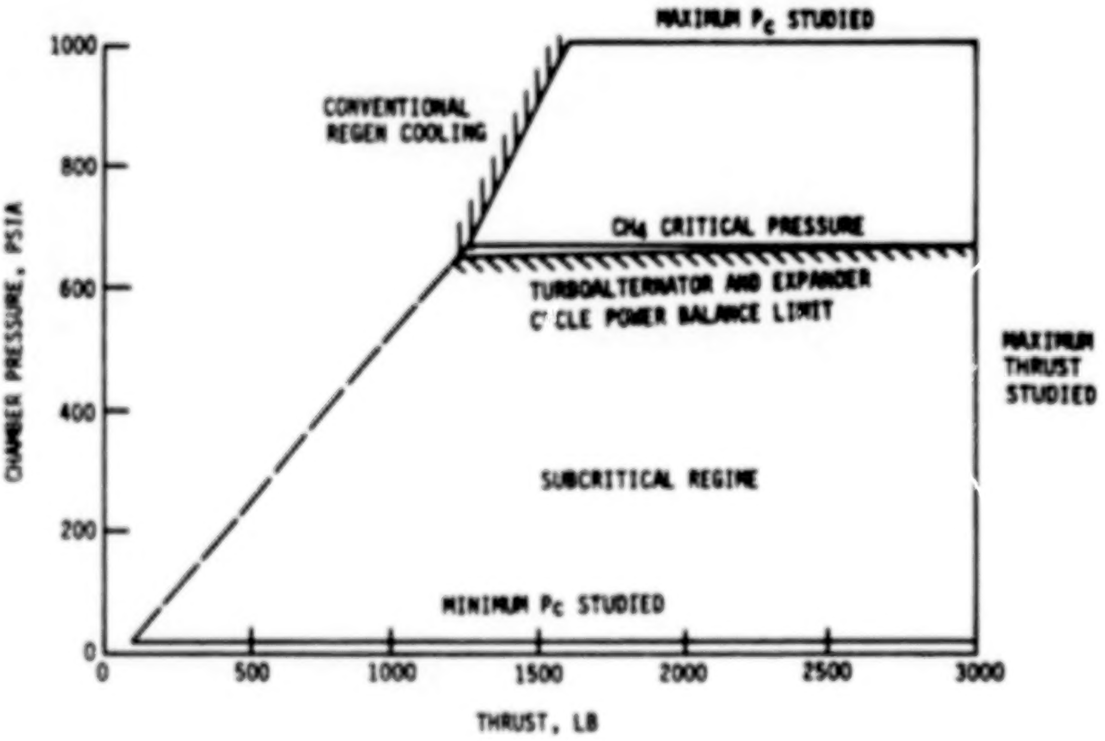
O_2/CH_4 Turboalternator and Expander Cycle Operating Range

This operating map is similar to that described for O_2/H_2 except the engine combined power balance and coolant limit occured at a lower thrust level (~1300 psia). In conducting these power balances, the coolant jacket exit pressure was maintained above the critical pressure of CH_4. This, of course, put the burden upon the methane pumping system.

O_2/CH_4 Turboalternator And Expander Cycle Operating Range

CH_4 REGEN COOLING

O/F = 3.7



Turboalternator Cycle Performance Parametrics

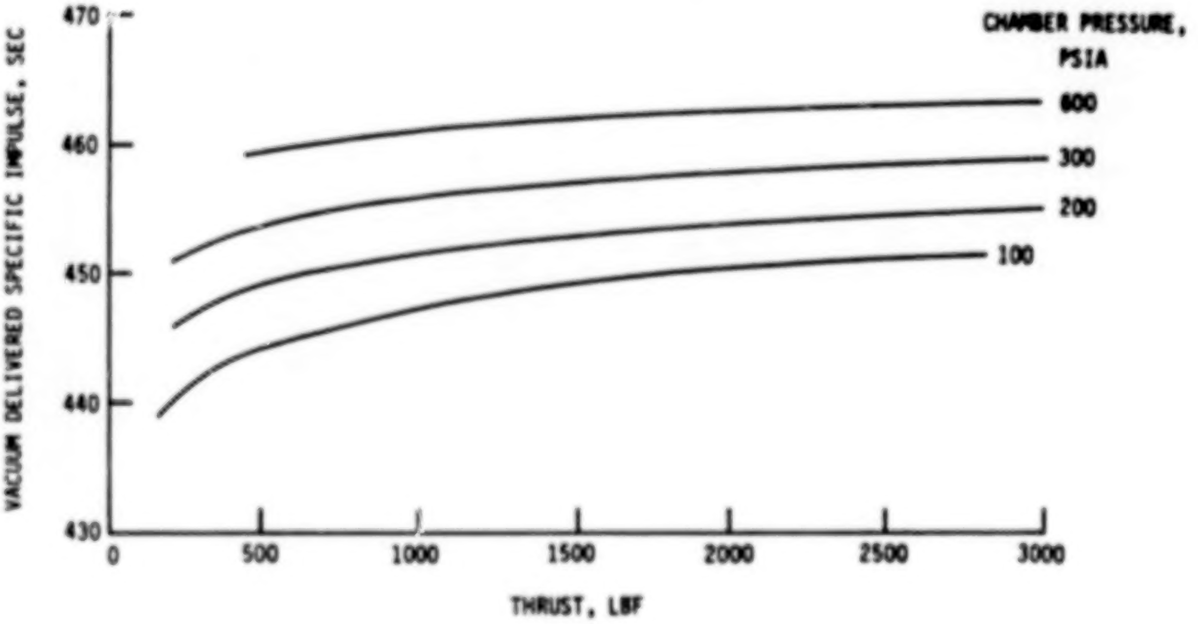
Typical parametric data generated by the study is shown on this and the following two charts. The engine delivered performance data for a turboalternator cycle is shown as a function of both thrust and chamber pressure. This data is also applicable for an O_2/H_2 expander cycle. Performance decreases with both decreasing thrust and chamber pressure because of the kinetic loss increases. The energy release efficiency also decreases with chamber pressure. With LO_2/GH_2 propellants, energy release efficiencies (ERE) greater than 98% can be achieved. An ERE of .995 at 1000 psia and .992 at 100 psia is considered typical of the state-of-the-art for LO_2/GH_2 propellants.

Turboalternator Cycle Performance Parametrics

O_2/H_2 PROPELLANTS

O/F = 6.0

ϵ = 400:1



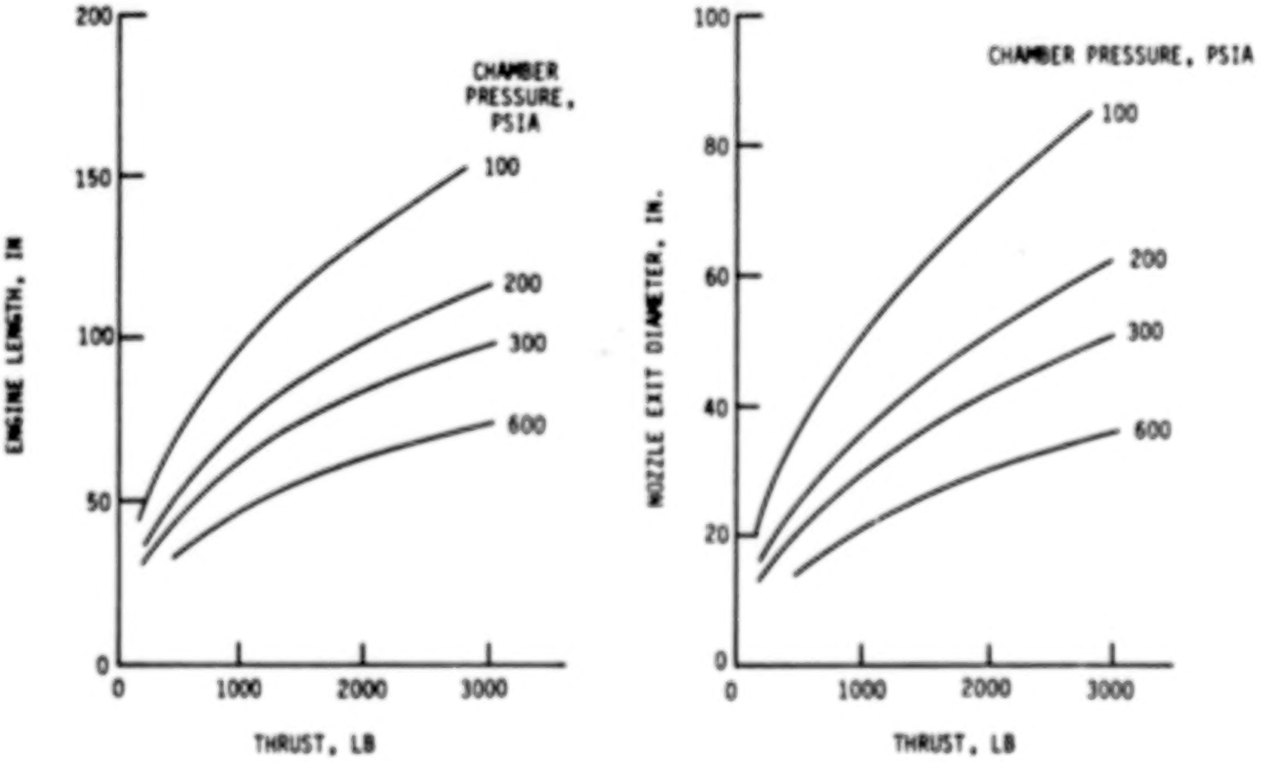
Turboalternator Cycle Envelope Parametrics

The engine envelope (total length and nozzle exit diameter) is shown as functions of thrust and chamber pressure for the turboalternator cycle. This data is also applicable to an expander cycle engine. The data shows that the nozzles get very large at high thrust and at low chamber pressures. This, of course, gets reflected in the engine weight and is discussed with the following chart.

If short engine lengths are required to minimize the length of the COTV, high chamber pressure, low thrust operation is desirable. This increases the incentive to look at advanced cooling schemes.

Turboalternator Cycle Envelope Parametrics

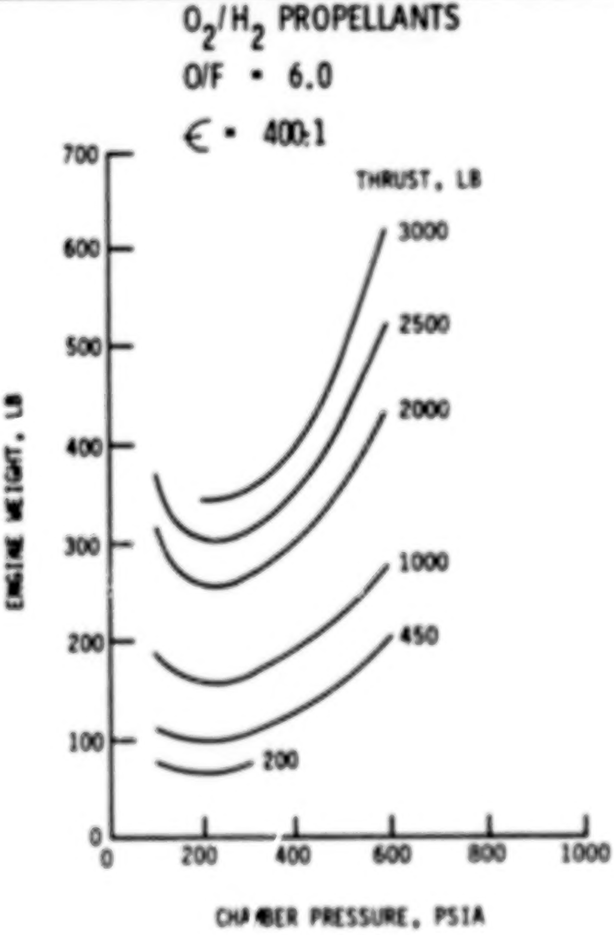
O_2/H_2 PROPELLANTS
O/F = 6.0
ϵ = 400:1



Turboalternator Cycle Weight Parametrics

The engine weight data for the turboalternator cycle is shown on the figure as a function of chamber pressure. Engine weight increases as chamber pressure decreases below 200 psia because the nozzle becomes the dominant component weight. As expected, this is amplified as the engine thrust increases. As thrust and chamber pressure increase, the alternator and electric motors become the dominant heavy components. This is particularly evident at a thrust level of 3000 lbs and a chamber pressure of 600 psia.

Turboalternator Cycle Weight Parametrics



Task IV Engine System Preliminary Design Objectives

Task IV takes the outputs of all previous task analyses and molds them into a final end product: the preliminary design of two engine concepts. One of these designs will be of an oxygen/hydrogen engine and the other of an oxygen/hydrocarbon engine.

Based upon the component design analyses, layouts, performance and cycle balance, baseline engine performance, weight and envelope data will be calculated for each of the two engine concepts at an area ratio of 400:1. This data will be used to modify the weight and envelope scaling equations and adjust the performance loss calculations in the parametric engine models. The parametric data generated for these two engine concepts will then be updated and the performance, weight and envelope data presented as functions of thrust, chamber pressure and area ratio. The thrust and chamber pressure ranges will be the feasible design ranges established in Task III and the area ratio range will be 200 to 1,000.

Task IV Engine System Preliminary Design Objectives

- PRELIMINARY DESIGN OF TWO ENGINE CONCEPTS
  - O_2/H_2
  - O_2/HYDROCARBON
- UPDATE ENGINE PARAMETRIC DATA
  - THRUST & P_c PER TASK II AND III RESULTS
  - ϵ = 200 TO 1000

Major Technology Requirements

During the course of the study, recommendations for advanced technology or further study efforts which would enhance the concepts will be identified. Those items which have been identified at this point in the study are shown on the figure.

Advanced cooling schemes are required if O_2/RP-1 engines are to become viable low-thrust candidates. If the performance losses associated with film cooled engines are too high and engine envelope is a system design driver, then advanced cooling schemes are required for all propellant combinations to increase the operating chamber pressures at low-thrust.

If a pump-fed system is selected for this application, the development of high efficiency pumps in this small size range is required because experience in certain sizes is either non-existent or very limited.

The 100 to 3000 lbf thrust range being studied is too broad because problems or design drivers may vary significantly in this range. Engine/vehicle study effort should be continued to better define the engine thrust requirement and to focus on the real issues.

Major Technology Requirements

- DEVELOP ADVANCED COOLING SCHEMES TO EXTEND REGENERATIVE COOLING LIMITS.
- DEVELOP LOW SPECIFIC SPEED, HIGH HEAD RISE, LOW FLOW, HIGH EFFICIENCY TURBOPUMPS.
- CONDUCT FURTHER EFFORT TO REDUCE THE THRUST AND CHAMBER PRESSURE RANGES.

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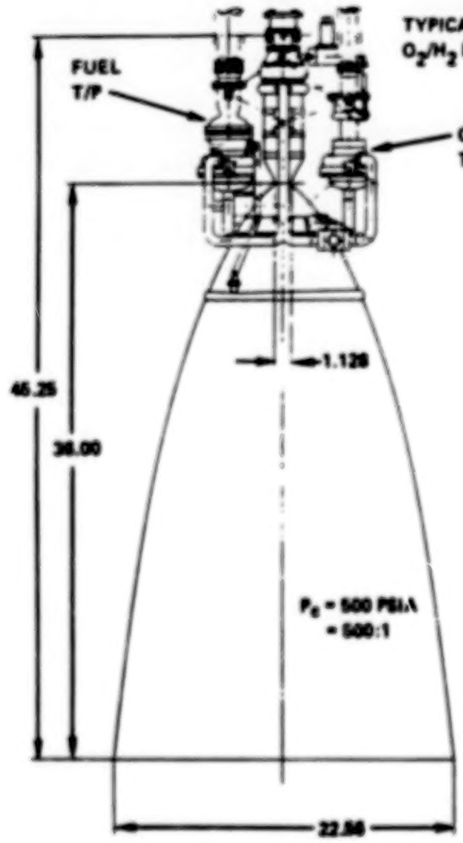
LOW-THRUST CHEMICAL PROPULSION

James M. Shoji
Rockwell International Corporation

This presentation will summarize the results of an on-going contract with NASA-LeRC. The NASA-LeRC Project Manager is Dean Scheer and the Rocketdyne Program Manager is Hal Diem. The results will include: (1) Thrust chamber cooling analysis and results; and (2) Engine cycle/configuration limits; and (3) Engine performance data.

This chart presents the basic objective, approach, and the desired results of the program. The primary program objective is to define low-thrust chemical engine concepts. The approach is to consider three candidate propellant combinations (O_2/H_2, O_2/CH_4, and O_2/RP-1) for both pump and pressure-fed engines with a thrust range of 100 lb. to 3000 lb. and a chamber pressure range of 20 to 1000 psia. The program results are to include a formulation of the propulsion system concept and a definition of required technology.

LOW THRUST CHEMICAL ROCKET ENGINE STUDY



TYPICAL 1000 LB THRUST ENGINE
O_2/H_2 EXPANDER CYCLE

OBJECTIVE

- DEFINE LOW-THRUST CHEMICAL ENGINE CONCEPTS

APPROACH

- O_2/H_2, O_2/CH_4, O_2/RP-1 PROPELLANTS
- PUMPED AND PRESSURE FED
- 100 TO 3000 LB THRUST RANGE
- 20 TO 1000 PSIA CHAMBER PRESSURE RANGE

RESULTS

- PROPULSION SYSTEM CONCEPT FORMULATION
- TECHNOLOGY PROGRAM DEFINITION

For the low thrust engine two conventional thrust chamber cooling techniques were to be evaluated. These were regenerative/radiation and film/radiation cooling which utilized the fuel as the coolant. With the three propellant combinations and the two cooling techniques, a total of six cases can be configured.

LOW THRUST RANGE OF INTEREST*

CASE NO.	PROPELLANTS	MIXTURE RATIO	COOLING METHOD	COOLANT	THRUST STUDY RANGE, POUNDS	CHAMBER PRESSURE STUDY RANGE, PSIA
1	O_2/H_2	6.0	REGEN	H_2	100 TO 3000	20 TO 1000
2	O_2/H_2	6.0	FILM	H_2	100 TO 3000	20 TO 1000
3	O_2/RP-1	3.0	REGEN	RP-1	100 TO 3000	20 TO 1000
4	O_2/RP-1	3.0	FILM	RP-1	100 TO 3000	20 TO 1000
5	O_2/CH_4	3.7	REGEN	CH_4	100 TO 3000	20 TO 1000
6	O_2/CH_4	4.7	FILM	CH_4	100 TO 3000	20 TO 1000

*FROM TABLE I OF THE RFP

This chart presents the analysis guidelines primarily associated with the thrust chamber cooling evaluation. A nozzle with a 400-to-1 area ratio and 90-percent length was specified for this portion of the study. Combustion chamber lengths and contraction ratios were sized to achieve a minimum combustion efficiency of 98-percent. The film/radiation-cooled thrust chambers were permitted a maximum of 10-percent cooling loss. For hydrocarbon fueled propellants, the benefit of the gas-side carbon layer was to be neglected although current add-on studies will evaluate its influence. For the regenerative/radiation-cooled thrust chambers, a milled-channel wall combustor using NARloy-Z ($Twg_{max} = 1000^{\circ}F$) or nickel ($Twg_{max} = 1300^{\circ}F$) was used. These temperature limits were set based on a hardware durability standpoint. The nozzle was to be a stainless steel tubular construction. For regenerative-cooling, the maximum coolant velocity and the coking temperature limits for the hydrocarbon fuels were specified as shown. Also the coolant flow within the thrust chamber must be stable. For film/radiation-cooling, conventional wall materials and their respective maximum temperature limits were used. The thrust chamber cycle life required was five thermal cycles times a safety factor of four.

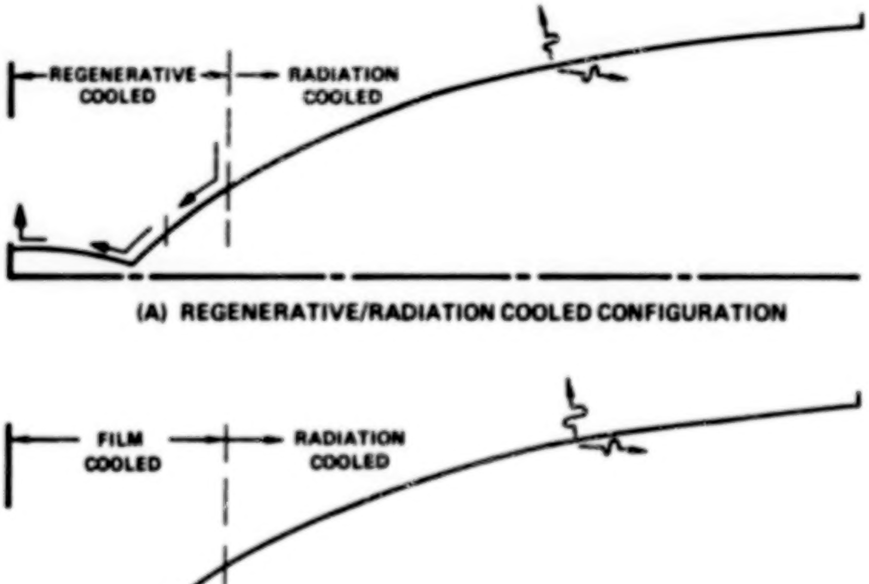
ANALYSIS GUIDELINES

- THRUST: 100 TO 3000 LB
- PROPELLANTS:
    - O_2/H_2 AT MR = 6
    - O_2/CH_4 AT MR = 3.7
    - O_2/RP-1 TO MR = 3.0
- THRUST CHAMBER GEOMETRY: 400-TO-1 AREA RATIO (90% LENGTH)
- CHAMBER PRESSURE: 20 TO 1000 PSIA
- PERFORMANCE:
    - 98% COMBUSTION EFFICIENCY
    - FILM/RADIATION COOLED
        - $(P_C)_{MAX}$: $(\eta_{I_s})_{FILM} = 0.90$
- THRUST CHAMBER COOLING:
    - HOT-GAS HEAT TRANSFER
        - NEGLECT CARBON LAYER BENEFIT
    - REGENERATIVE/RADIATION
        - MATERIAL
            - COMBUSTOR: CHANNEL WALL
                - COPPER ALLOY: $(T_{WG})_{MAX} = 1000$ F
                - NICKEL: $(T_{WG})_{MAX} = 1300$ F
            - NOZZLE: TUBULAR
        - COOLANT (FUEL)
            - MAXIMUM COOLANT VELOCITY (REGENERATIVE-COOLED)
                - GAS: MACH NO. = 0.3
                - LIQUID: 200 FT/SEC
            - COKING LIMIT (REGENERATIVE-COOLED)
                - RP-1: $(T_{WC})_{MAX} = 560$ F
                - CH_4: $(T_{WC})_{MAX} = 1300$ F
            - COOLANT FLOW MUST BE STABLE
    - FILM/RADIATION
        - MATERIAL
            - L605: $(T_{WG})_{MAX} = 2000$ F
            - MOLY: $(T_{WG})_{MAX} = 2500$ F
- CYCLE LIFE:
    - FIVE THERMAL CYCLES TIMES A SAFETY FACTOR OF FOUR
    - ACCUMULATIVE RUN TIME (FUNCTION OF THRUST)

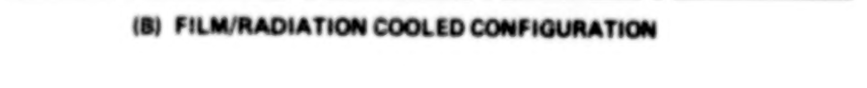
- GENERAL
    - STRUCTURAL
        - YIELD SAFETY FACTOR = 1.1
        - ULTIMATE SAFETY FACTOR = 1.4

This chart presents the two candidate thrust chamber cooling methods evaluated. The regenerative/radiation-cooled thrust chamber had a portion of nozzle and the combustion chamber regeneratively-cooled and the remainder of the nozzle was radiation cooled. The film/radiation-cooled thrust chamber had the film coolant injected at the injector face.

CANDIDATE THRUST CHAMBER COOLING METHODS


REGENERATIVE COOLED
RADIATION COOLED

(A) REGENERATIVE/RADIATION COOLED CONFIGURATION


FILM COOLED
RADIATION COOLED

(B) FILM/RADIATION COOLED CONFIGURATION

The method of analysis for the radiation-cooled portion of the nozzle utilized an integral boundary layer computer program with conventional wall materials to determine the nozzle wall temperature profile and define parametric nozzle attach area ratio data. For regenerative-cooling the gas-side heat transfer coefficient distribution was determined utilizing a combination of the integral boundary layer computer program results and extrapolated test data. The test data are used to provide a more realistic distribution near the injector. The coolant-side heat transfer coefficient was determined using existing coolant correlations. For example, for hydrogen the modified Dipprey -Sabersky coolant correlation was used. For methane a generalized coolant correlation was assumed; and for RP-1, the coolant correlation developed from the F-1 and Atlas Program was used. The thrust chamber coolant passage design utilized the regenerative-cooling design/analysis computer program. This computer program is capable of both design and analysis of channel wall or tubular coolant passages and is capable of performing two-dimensional wall temperature calculations as well as structural analysis of the coolant passage and predicts thrust chamber cycle life.

THRUST CHAMBER COOLING: ANALYSIS APPROACH

- RADIATION COOLING
  - METHOD OF ANALYSIS
    - ROCKETDYNE INTEGRAL BOUNDARY LAYER COMPUTER PROGRAM
  - CONVENTIONAL WALL MATERIALS
    - L605
    - MOLYBDENUM WITH OXIDATION PROTECTION COATING
  - DETERMINE WALL TEMPERATURE PROFILE
    - DEFINE NOZZLE ATTACH AREA RATIO
- REGENERATIVE-COOLING
  - METHOD OF ANALYSIS
    - GAS-SIDE HEAT TRANSFER COEFFICIENT
      - ROCKETDYNE INTEGRAL BOUNDARY LAYER COMPUTER PROGRAM
      - EXTRAPOLATED TEST DATA
    - COOLANT-SIDE HEAT TRANSFER COEFFICIENT
      - EXISTING COOLANT CORRELATIONS
    - COOLANT PASSAGE DESIGN
      - ROCKETDYNE REGENERATIVE-COOLING DESIGN/ANALYSIS COMPUTER PROGRAM

The wall materials considered for regenerative-cooling included NARloy-Z, cres, and nickel. The regenerative-cooling analysis defined the cooling limits based on the analysis guidelines, determined coolant passage design, and provided parametric data on thrust chamber coolant heat input and coolant pressure drop.

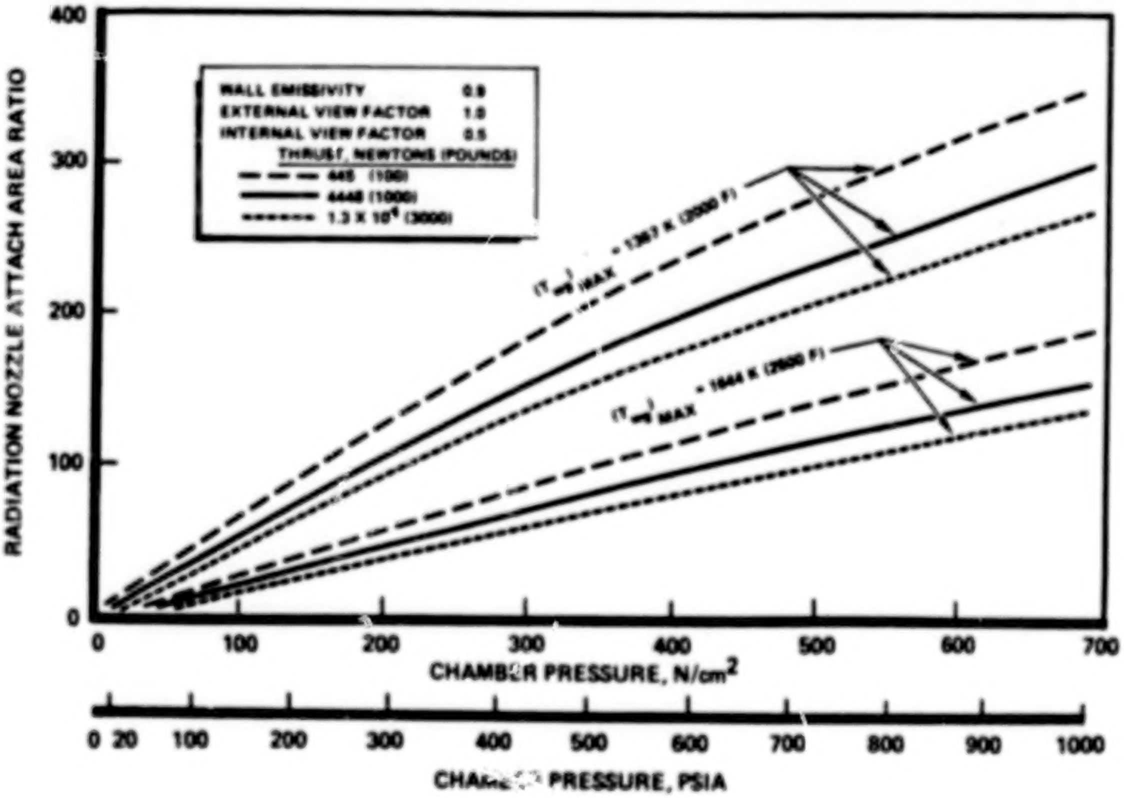
For film cooling, the linear mixture ratio profile model (simplified JANNAF analysis approach) was utilized to determine the maximum allowable film-coolant flow (10-percent cooling loss). The thrust chamber film-cooling heat transfer analysis to obtain wall temperatures and cooling limits utilized a gaseous film-cooling model for supercritical pressures and a liquid film-cooling model for subcritical pressures.

THRUST CHAMBER COOLING: ANALYSIS APPROACH

- •REGENERATIVE-COOLING
    - •NARLOY-Z, CRES AND/OR NICKEL
    - •HEAT TRANSFER DATA
        - •DEFINE COOLING LIMITS
        - •DETERMINE COOLANT PASSAGE DESIGN
        - •DETERMINE COOLANT HEAT INPUT AND COOLANT PRESSURE DROP
- •FILM-COOLING
    - •METHOD OF ANALYSIS
        - •LINEAR MR PROFILE FILM COOLING MODEL
        - •ROCKETDYNE GASEOUS AND LIQUID FILM-COOLING COMPUTER PROGRAMS
    - •WALL MATERIALS
        - •L605 OR MOLYBDENUM WITH OXIDATION PROTECTION COATING
    - •HEAT TRANSFER DATA
        - •DETERMINE REQUIRED COOLANT FLOW
        - •DEFINE COOLING LIMITS

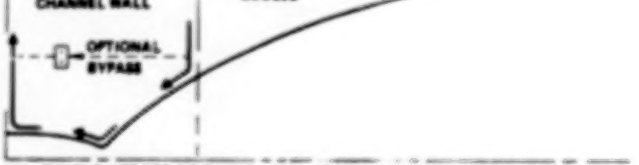
This chart presents the results of the radiation-cooled nozzle analysis for O_2/H_2. Radiation nozzle attach area ratios for two maximum wall temperatures (2000°F and 2500°F) are presented for thrust levels of 100, 1000, and 3000 lbs. Results of a preliminary in-house design effort indicated that for a retractable nozzle (to achieve a reduced engine length), a convenient cutoff area ratio was approximately 200-to-1 area ratio. If this value is selected, all O_2/H_2 thrust chambers in the thrust and chamber pressure range of interest will have a maximum wall temperature less than 2500°F for the radiation-cooled portion of the nozzle. Also since O_2/H_2 is the most energetic of the three propellant combinations, the radiation-cooled nozzle wall temperatures would even be lower for O_2/CH_4 and O_2/RP-1.

RADIATION NOZZLE ATTACH AREA RATIO VARIATION WITH CHAMBER PRESSURE AND THRUST FOR LO_2/H_2

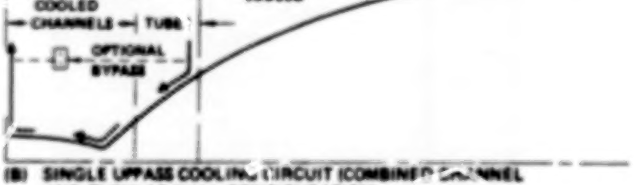


For the regenerative/radiation-cooled thrust chamber, four regenerative cooling circuits were initially evaluated. Cooling circuits A and B are single uppass circuits. Circuit C is a split-flow cooling circuit in which the coolant flows through the combustor and nozzle in parallel. The series cooling circuit (Circuit D) was selected as the baseline due to its lower coolant pressure drop for the low thrust conditions of interest.

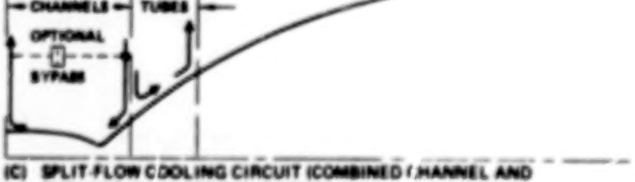
TYPICAL REGENERATIVE COOLING CIRCUITS


REGENERATIVELY
COOLED
CHANNEL WALL
RADIATION
COOLED
OPTIONAL
BYPASS

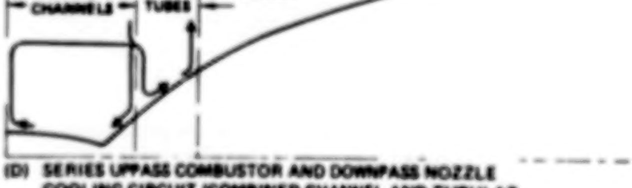
(A) SINGLE UPPASS COOLING CIRCUIT (ALL CHANNEL WALL)


REGENERATIVELY
COOLED
CHANNELS
TUBES
RADIATION
COOLED
OPTIONAL
BYPASS

(B) SINGLE UPPASS COOLING CIRCUIT (COMBINED CHANNEL AND TUBULAR WALL CONFIGURATION)


REGENERATIVELY
COOLED
CHANNELS
TUBES
RADIATION
COOLED
OPTIONAL
BYPASS

(C) SPLIT-FLOW COOLING CIRCUIT (COMBINED CHANNEL AND TUBULAR WALL CONFIGURATION)

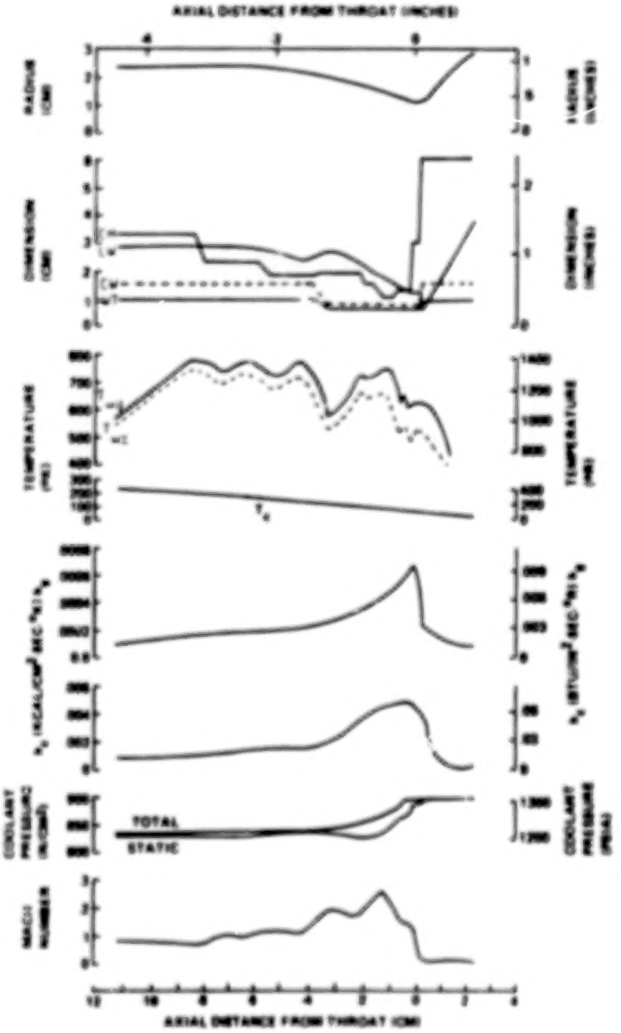

REGENERATIVELY
COOLED
CHANNELS
TUBES
RADIATION
COOLED

(D) SERIES UPPASS COMBUSTOR AND DOWNPASS NOZZLE COOLING CIRCUIT (COMBINED CHANNEL AND TUBULAR WALL CONSTRUCTION)

Detailed regenerative-cooled thrust chamber analyses were performed for a discrete number of cases to define the cooling limits and obtain heat transfer data for input into the engine cycle analysis. This chart presents the detail analysis results for a typical LO_2/H_2 combustor (injector to a low supersonic area ratio). The design condition was 1000 LBf thrust and a chamber pressure of 1000 psia at a mixture ratio of 6.0. The combustor contour along with coolant channel dimensions, wall temperatures (two-dimensional), gas-side and coolant-side film coefficients, coolant pressures and coolant Mach number distributions are presented. As noted in this chart, the maximum wall temperature is below the 1460°R maximum allowable for NARloy-Z and the coolant Mach number is slightly below the maximum allowable of 0.3. Therefore this condition represents a thrust chamber on the regenerative-cooling limit.

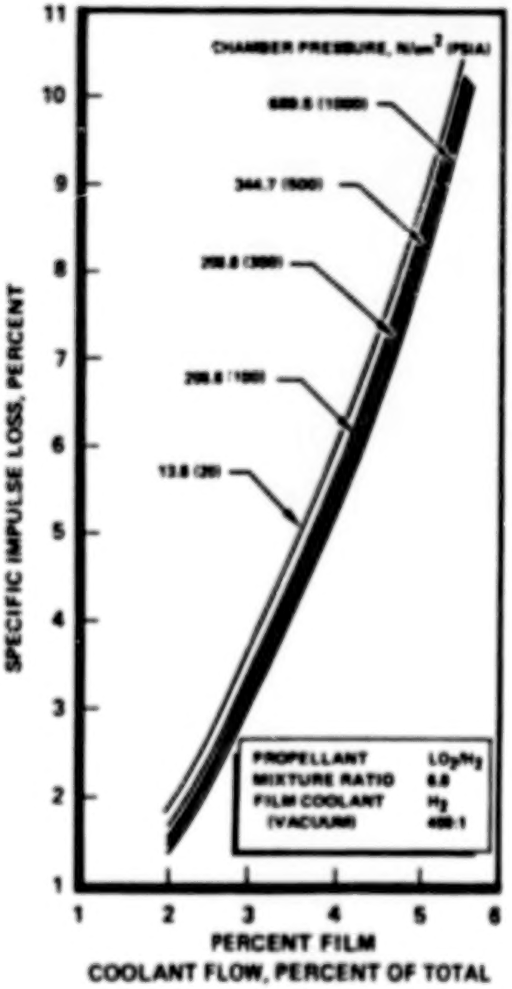
PARAMETERS FOR THE O_2/H_2 LOW THRUST COMBUSTOR

MIXTURE RATIO	6.0
THRUST, NEWTONS (LBF)	4448 (1000)
CHAMBER PRESSURE, N/CM² (PSIA)	689.5 (1000)



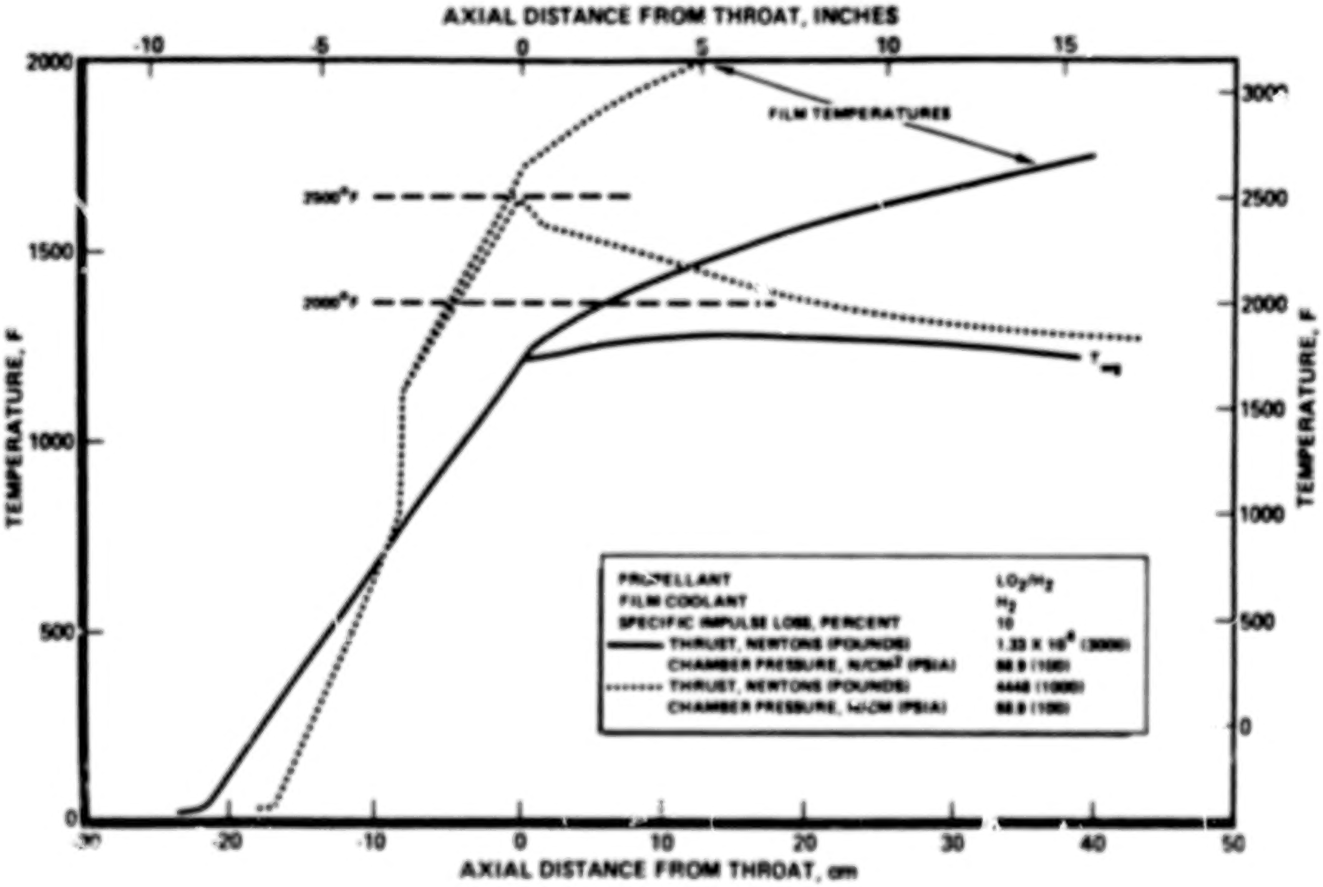
For the film/radiation-cooled thrust chamber, the maximum allowable film coolant flowrate was determined by using the linear mixture ratio profile film cooling performance loss model. For the maximum 10-percent performance loss (see Study Guidelines), a film coolant flow of approximately 5.5-percent resulted for LO_2/H_2 with a nozzle area ratio of 400-to-1. Also note that the resulting film coolant flow was rather insensitive to chamber pressure.

LO_2/H_2 FILM COOLING PERFORMANCE LOSS



Using these allowable film coolant flowrates, detailed heat transfer analyses were performed for a number of design conditions to define the film/radiation-cooled thrust chamber cooling limits. Two typical analysis results are presented in this chart for LO_2/H_2 at a chamber pressure of 100 psia. Axial film and wall temperature distributions are shown. The lower thrust (1000 LB_f) resulted in a higher wall temperature (approximately 2500°F) due to the lower hydraulic diameter causing higher heat fluxes. The deviation of the film and the wall temperature downstream of the throat is due to radiation-cooling. For a maximum allowable temperature of 2500°F, the 1000 LB_f thrust design condition is on the cooling limit for the film/radiation-cooled thrust chamber.

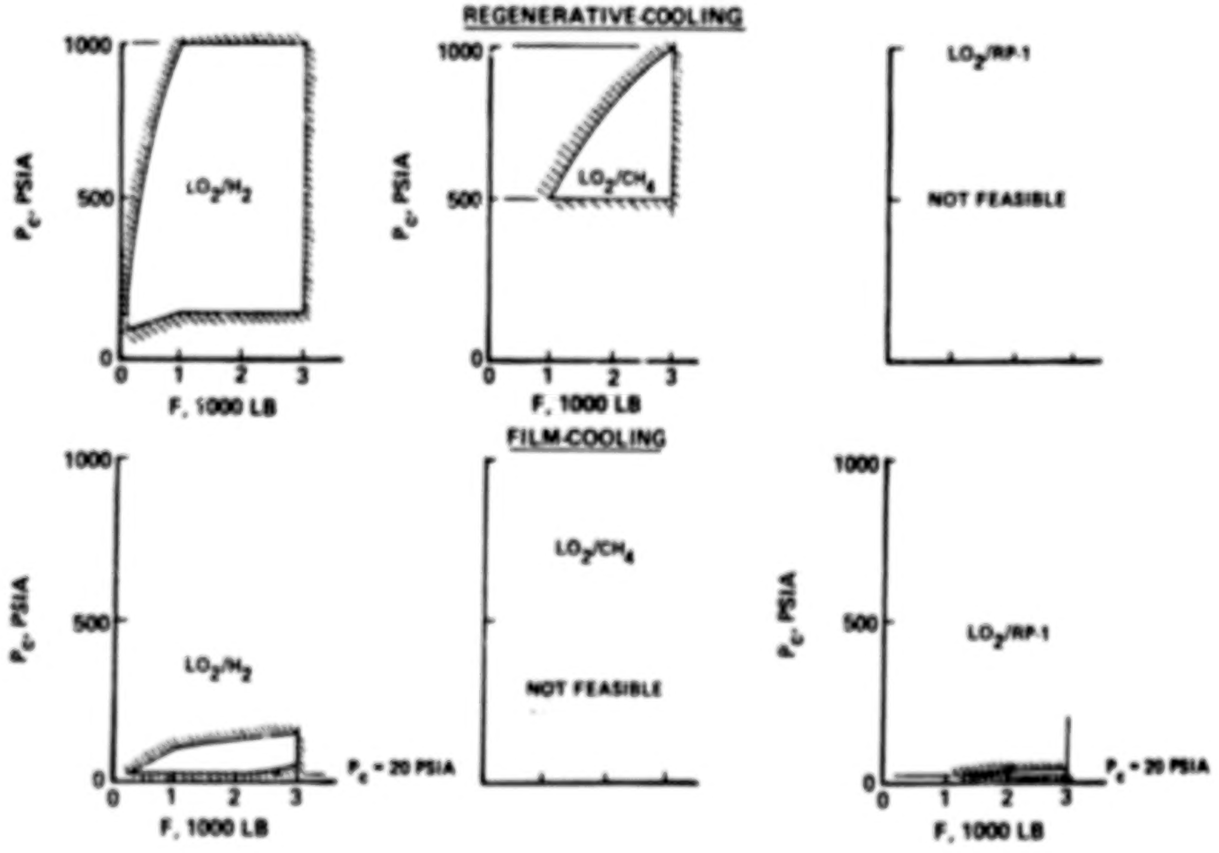
LO_2/H_2 FILM-COOLED THRUST CHAMBER RESULTS



This chart presents a summary of the thrust chamber cooling limits for both regenerative and film cooling. Above 1000 LB thrust, the LO_2/H_2 regenerative-cooled thrust chamber maximum chamber pressures exceeded the maximum study chamber pressure of 1000 psia; however, below 1000 LB thrust the maximum chamber pressure decreased to 200 psia at 100 LB thrust. The minimum chamber pressure was set to maintain a coolant pressure above the critical pressure due to coolant flow instability resulting from two-phase flow. For LO_2/CH_4 the operational envelope was considerably less for LO_2/H_2 due to the poorer cooling capability of Methane and higher critical pressure. Regenerative-cooling for LO_2/RP-1 was found to be not feasible, primarily the result of neglecting the gas-side carbon layer. This influence will be evaluated as part of the program add-on effort.

The operational envelopes for film cooling were limited to a maximum chamber pressure of approximately 150 psia which was for LO_2/H_2. The LO_2/CH_4 film-cooled thrust chambers were found to be not feasible and the operational envelope for LO_2/RP-1 thrust chambers was extremely limited.

THRUST CHAMBER COOLING LIMIT SUMMARY



The engine cycle/configuration analyses approach consisted of first a definition of candidate cycles including the work statement specified configurations and the incorporation of the heat transfer analysis results. The analyses of the resultant engine cycle/configurations was performed using the Rocketdyne Low Thrust Engine Cycle Balance Computer Program which is capable of simultaneously optimizing up to eight parameters. The alternator, electric motor, and fuel cell data and design relationships were incorporated in the computer program. These analyses defined the engine cycle limits (maximum design chamber pressure) and provided the engine balance data. Parametric thrust chamber performance data were also generated.

Currently the screening and evaluation of the engine cycle/configurations are being performed by determining the cycle operational capability, performance, envelope, weight, complexity, and technology advancement required.

# ENGINE CYCLE/CONFIGURATION EVALUATION: ANALYSIS APPROACH

- ENGINE CYCLE/CONFIGURATION DEFINITION AND MATRIX REFINEMENT
  - WORK STATEMENT SPECIFIED CONFIGURATIONS
  - INCORPORATION OF HEAT TRANSFER ANALYSIS RESULTS
- ENGINE CYCLE/CONFIGURATION ANALYSIS
  - METHOD OF ANALYSIS
    - ROCKETDYNE LOW THRUST ENGINE CYCLE BALANCE COMPUTER PROGRAM
      - INCORPORATION OF ALTERNATOR, ELECTRIC MOTOR, AND FUEL CELL DATA AND DESIGN RELATIONSHIPS
  - DETERMINE PARAMETRIC THRUST CHAMBER PERFORMANCE DATA
  - DEFINE ENGINE CYCLE LIMITS
    - ENGINE BALANCE DATA
- ENGINE CYCLE/CONFIGURATION SCREENING EVALUATION AND SELECTION
  - CYCLE OPERATIONAL CAPABILITIES
  - PERFORMANCE
  - ENVELOPE
  - WEIGHT
  - COMPLEXITY
  - TECHNOLOGY ADVANCES REQUIRED

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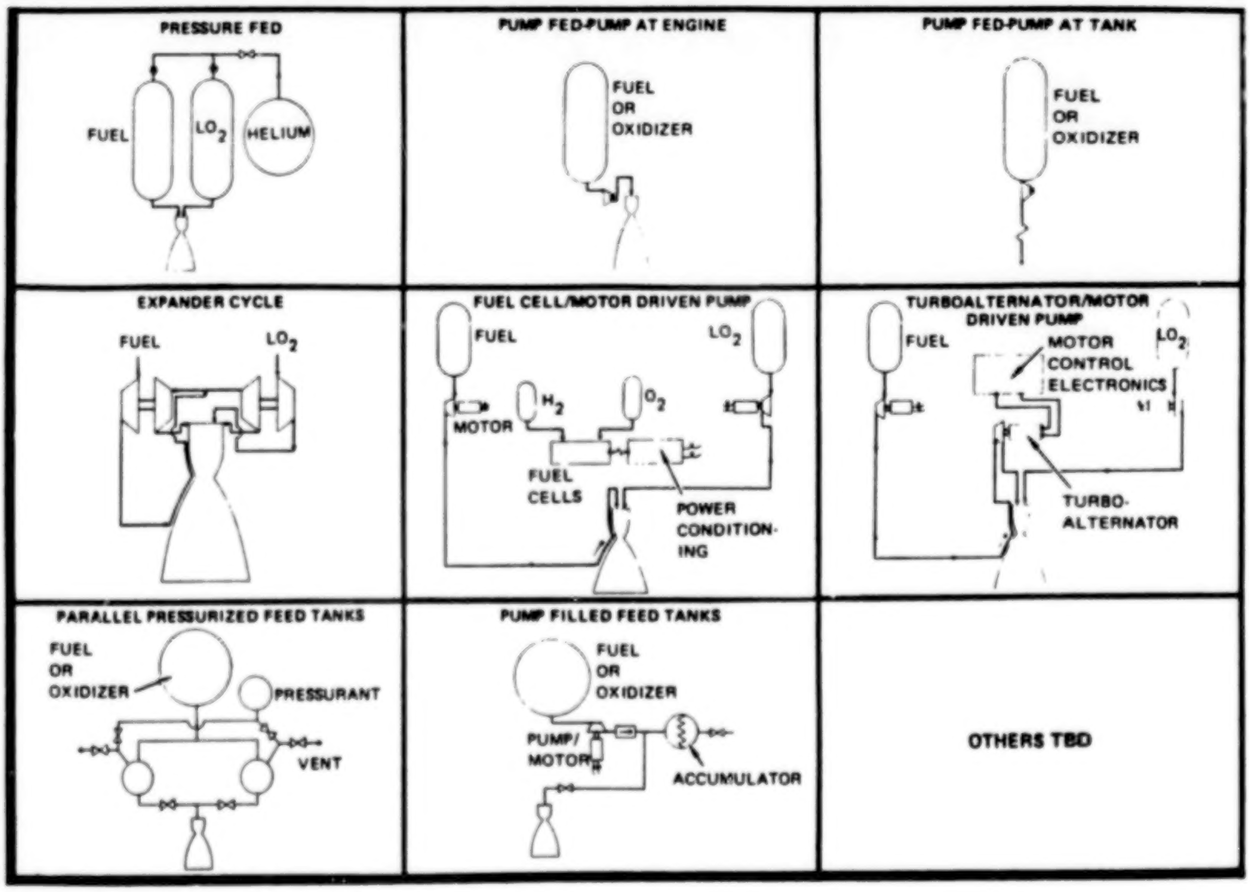
This chart schematically illustrates the candidate engine cycle/configurations. The engines include both pressure-fed and pump-fed engines. The pump-fed engines have the pumps located on the engine or at the tank. Conventional gas driven turbine cycles such as the direct expander cycle are candidates as well as unconventional cycles such as the fuel cell/motor driven pump cycle, turboalternator cycles, parallel pressurized feed tank, and pump-filled tank cycle.

ENGINE SYSTEM CONCEPTS TO BE STUDIED

(O_2/H_2, O_2/RP-1, O_2/CH_4 PROPELLANTS; REGEN. AND FILM COOLING)


PRESSURE FED
FUEL
LO_2
HELIUM
PUMP FED-PUMP AT ENGINE
FUEL
OR
OXIDIZER
PUMP FED-PUMP AT TANK
FUEL
OR
OXIDIZER
EXPANDER CYCLE
FUEL
LO_2
FUEL CELL/MOTOR DRIVEN PUMP
FUEL
LO_2
H_2
O_2
MOTOR
FUEL
CELLS
POWER
CONDITION-
ING
TURBOALTERNATOR/MOTOR
DRIVEN PUMP
FUEL
MOTOR
CONTROL
ELECTRONICS
LO_2
TURBO-
ALTERNATOR
PARALLEL PRESSURIZED FEED TANKS
FUEL
OR
OXIDIZER
PRESSURANT
VENT
PUMP FILLED FEED TANKS
FUEL
OR
OXIDIZER
PUMP/
MOTOR
ACCUMULATOR
OTHERS TBD

The resulting engine cycle/configuration matrix for the three propellant combinations and two cooling approaches is presented in this chart. The open boxes indicate the candidate engine cycles and the shaded boxes depict cycles which have been eliminated due to technical unfeasibility noted in the chart. Majority of eliminations occurred as a result of the incorporation of heat transfer results.

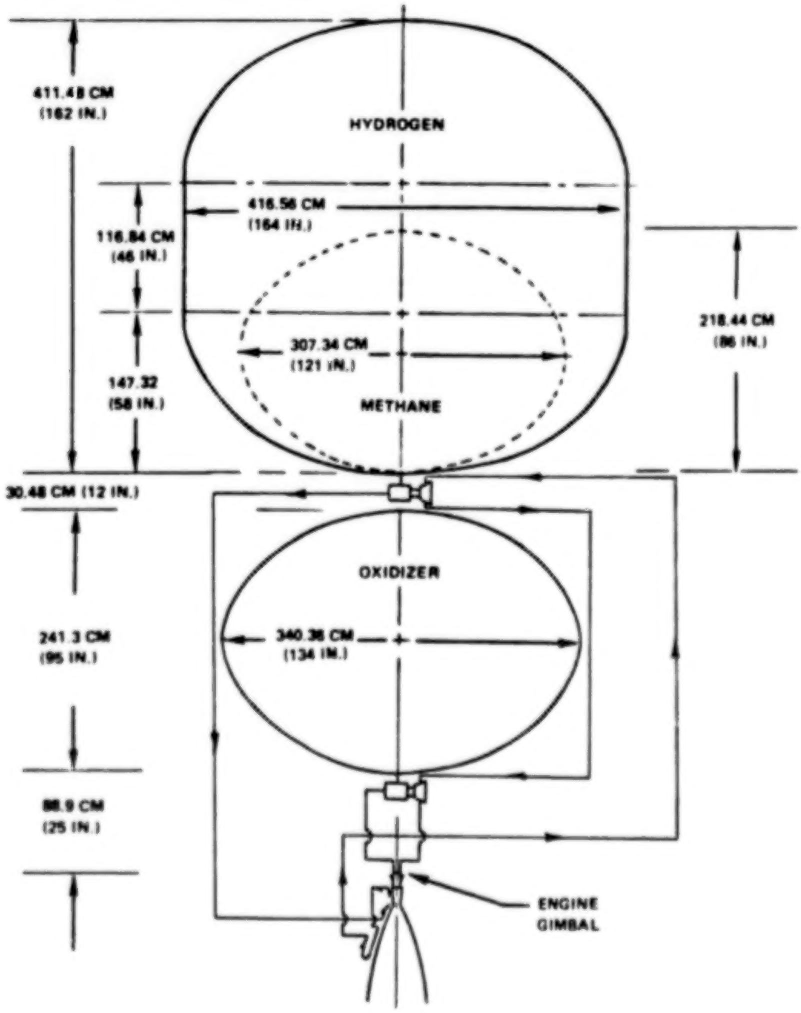
ENGINE CONFIGURATION MATRIX

PROPELLANT	O_2/H_2 (MR = 6.0)		O_2/CH_4 (MR = 3.7)		O_2/RP-1 (MR = 3.0)	
COOLING	REGEN COOLED	FILM COOLED	REGEN COOLED	FILM COOLED	REGEN COOLED	FILM COOLED
ENGINE MOUNTED PUMP FED						
EXPANDER CYCLE		▨ (1)		▨ (1)	▨ (2)	▨ (1)
GAS GENERATOR CYCLE				▨ (4)	▨ (3)	▨ (5)
STAGED COMBUSTION CYCLE				▨ (4)	▨ (3)	▨ (5)
TANK MOUNTED PUMP FED DIRECTLY POWERED PUMPS						
EXPANDER CYCLE		▨ (1)		▨ (1)	▨ (2)	▨ (1)
GAS GENERATOR CYCLE				▨ (4)	▨ (3)	▨ (5)
STAGED COMBUSTION CYCLE				▨ (4)	▨ (3)	▨ (5)
TANK-MOUNTED PUMP-FED INDIRECTLY POWERED PUMPS TURBO ALTERNATOR (WITH OR WITHOUT PUMP-FILLED FEED TANK)						
EXPANDER CYCLE		▨ (1)		▨ (1)	▨ (2)	▨ (1)
GAS GENERATOR CYCLE				▨ (4)	▨ (3)	▨ (5)
STAGED COMBUSTION CYCLE				▨ (4)	▨ (3)	▨ (5)
O_2/H_2 FUEL CELL POWERED SYSTEM				▨ (4)	▨ (3)	▨ (5)
PRESSURE-FED						
CONVENTIONAL			▨ (6)	▨ (4)	▨ (3)	
PARALLEL PRESSURIZED TANK			▨ (6)	▨ (4)	▨ (3)	

NOTES
(1) EXPANDER CYCLE REQUIRES HEATED PROPELLANT TO DRIVE TURBINES
(2) RP-1 EXPANDER CYCLE NOT FEASIBLE DUE TO COKING
(3) RP-1 REGENERATIVE COOLING NOT FEASIBLE } FROM HEAT TRANSFER RESULTS
(4) CH_4 FILM COOLING NOT FEASIBLE } FROM HEAT TRANSFER RESULTS
(5) MAXIMUM P_c (~25 PSIA) TOO LOW FOR PUMP FED LO_2/RP-1 ENGINES
(6) 500 PSIA CHAMBER PRESSURE TOO HIGH FOR PRESSURE FED LO_2/CH_4 ENGINE

For the tank-mounted pump/turbine engine cycles, the NASA-LeRC specified propellant tank configurations are illustrated. Both LO_2/H_2 and LO_2/CH_4 tank configurations are presented. An expander cycle with tank-mounted pumps and turbine is shown. These tank configurations enable the calculation of line lengths.

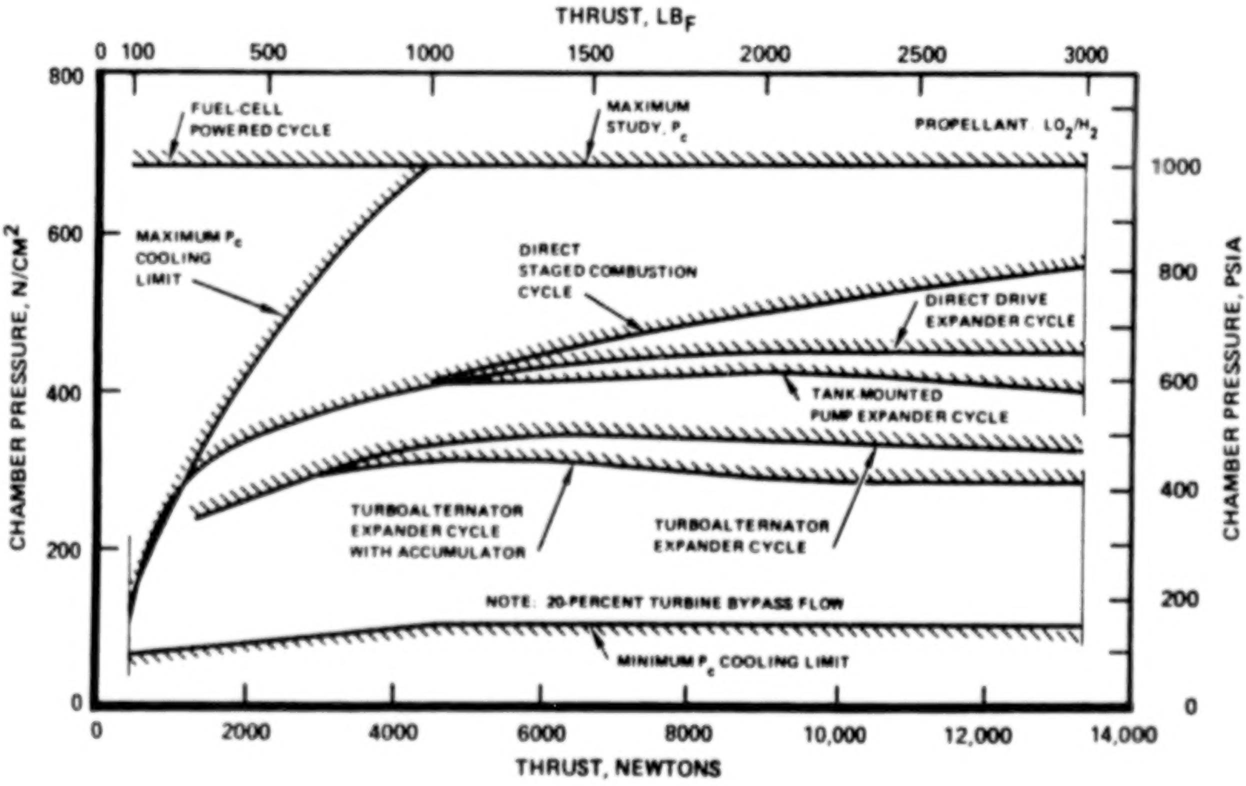
PROPELLANT TANK CONFIGURATIONS


411.48 CM
(162 IN.)
HYDROGEN
416.56 CM
(164 IN.)
116.84 CM
(46 IN.)
218.44 CM
(86 IN.)
307.34 CM
(121 IN.)
147.32
(58 IN.)
METHANE
30.48 CM (12 IN.)
OXIDIZER
241.3 CM
(95 IN.)
340.36 CM
(134 IN.)
88.9 CM
(25 IN.)
ENGINE
GIMBAL

This chart presents the regenerative-cooling and cycle limits for LO_2/H_2 engines. The fuel-cell powered cycle was capable of achieving the maximum study chamber pressure of 1000 psia for any thrust due to an almost unlimited available power. Whatever power was required to drive the pumps, a bigger fuel cell was incorporated. As a result the fuel cell system weight was, in general, an order of magnitude higher than the other engine concepts. The direct staged combustion cycle achieved the next highest chamber pressure; however, this cycle resulted in a marginal combustion stability for the preburners which could be detrimental.

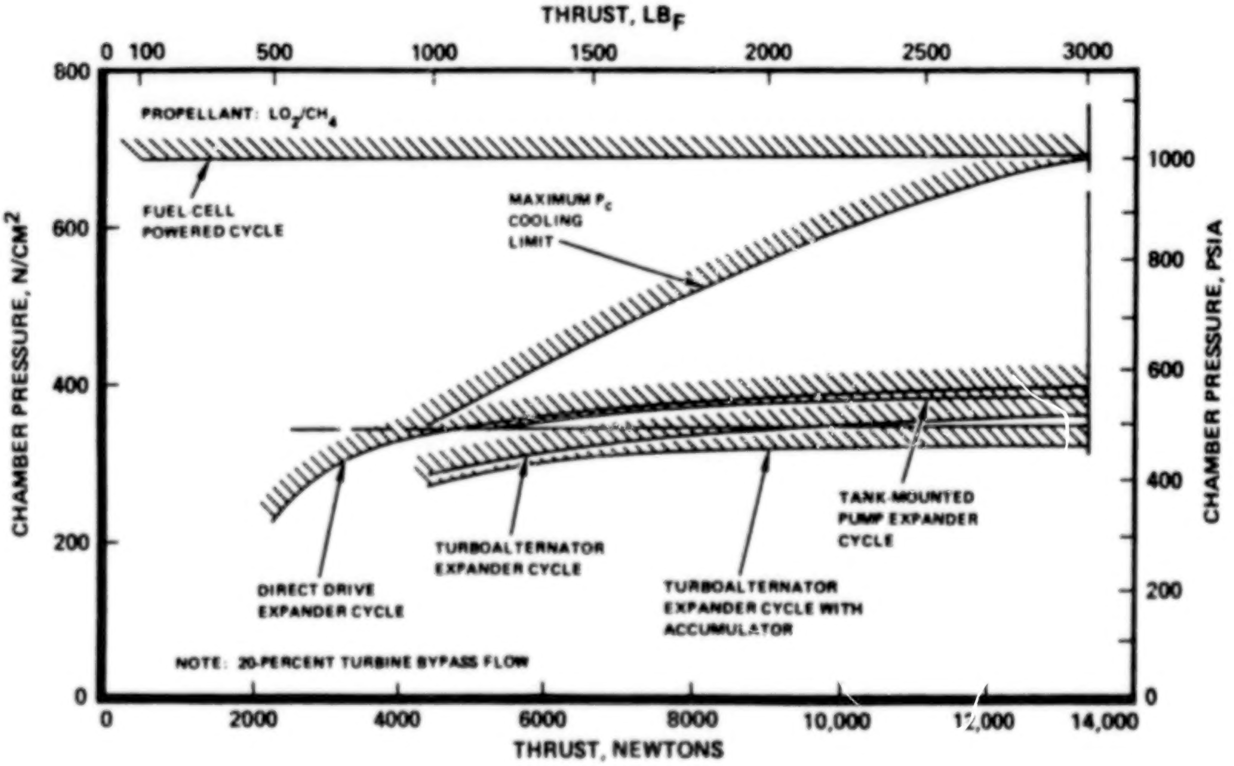
The next highest chamber pressure was achieved by the direct drive expander cycle. This cycle achieved a maximum chamber pressure of approximately 650 psia which remains essentially constant with decrease in thrust until 1000 LB. Modifications to the expander cycle all lead to a decrease in maximum chamber pressure at a given thrust. The tank-mounted pump expander cycle resulted in a lower maximum chamber pressure due to the additional pressure drop of the long hot-gas ducts. The inefficiencies of the added components (alternator and electric motors) decreased the maximum chamber pressure of the turboalternator expander cycle. The addition of the accumulator (pump-filled feed tank) improved the pump efficiencies but due to the increased propellant flow, required an increase in horsepower and therefore a decrease in chamber pressure resulted.

REGENERATIVE-COOLING AND CYCLE LIMITS FOR LO_2/H_2 ENGINES



Similar results occurred for the regenerative-cooled LO_2/CH_4 engines although the cycle limits were not as sensitive as for the LO_2/H_2 engines. Current analyses efforts indicate that the minimum chamber pressure limit for LO_2/CH_4 regenerative-cooling may be lower due to the increase in the actual coolant discharge pressure as a result of the turbine pressure ratios.

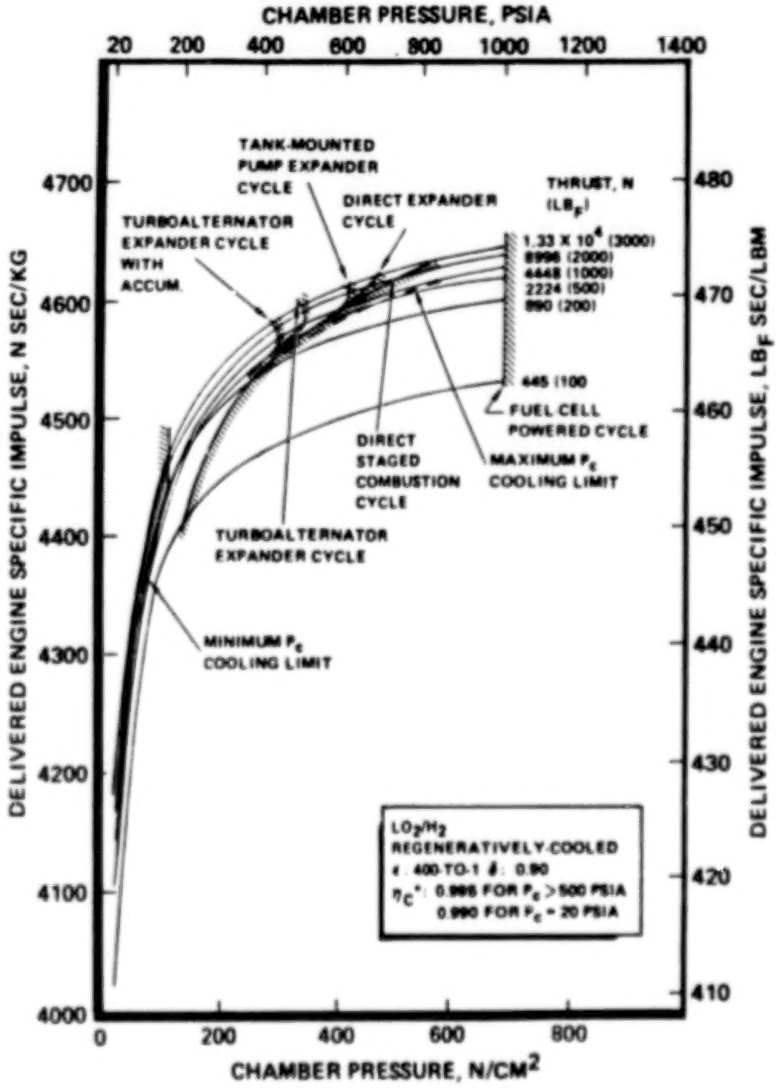
REGENERATIVE-COOLING AND CYCLE LIMITS FOR LO_2/CH_4 ENGINES



Parametric delivered engine specific impulse data are shown in this chart for regenerative-cooled LO_2/H_2 engines with both the cooling and cycle limits superimposed; and therefore clearly shows the maximum attainable engine specific impulse. These curves also show the rapid decrease in specific impulse below approximately 400 psia chamber pressure. Delivered specific impulses for the direct expander cycle engine can exceed 470 LB_f sec/LB_m.

LO_2/H_2 REGENERATIVELY-COOLED ENGINE DELIVERED SPECIFIC IMPULSE

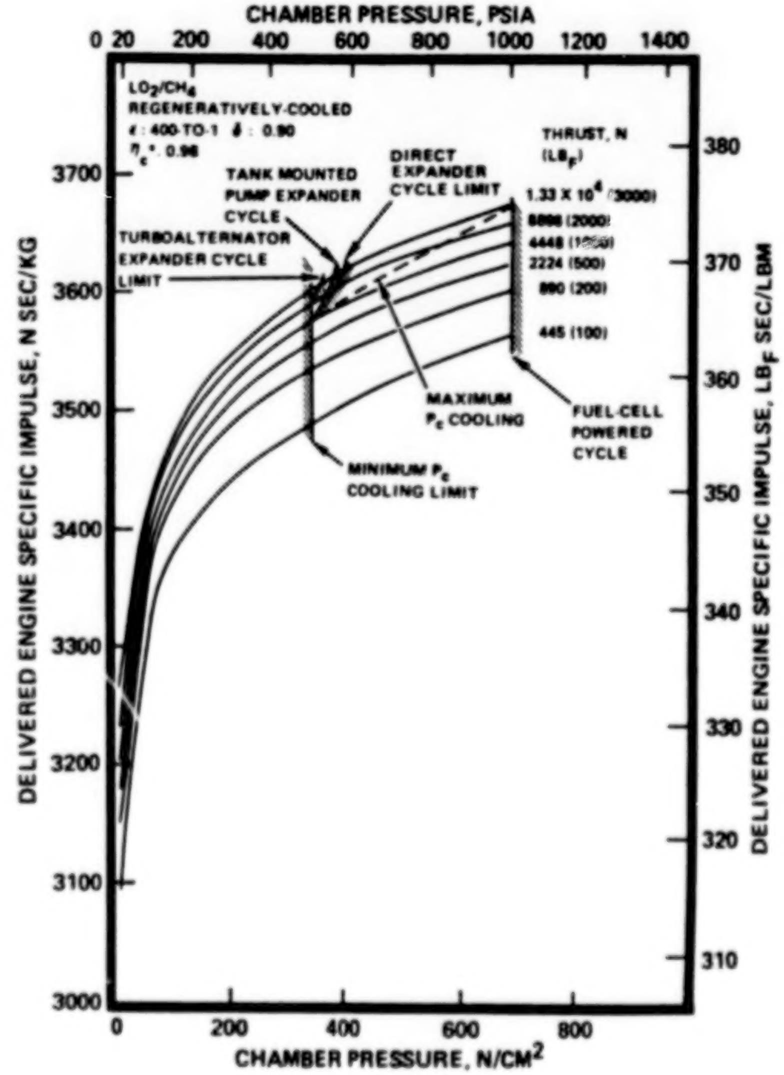
COOLING AND CYCLE LIMITS SUPERIMPOSED



Similar results for regenerative-cooled LO_2/CH_4 engines are presented in this chart. Delivered engine specific impulses are approximately 100 -LB_f sec/LB lower than these for the LO_2/H_2 engines.

LO_2/CH_4 REGENERATIVELY-COOLED ENGINE DELIVERED SPECIFIC IMPULSE

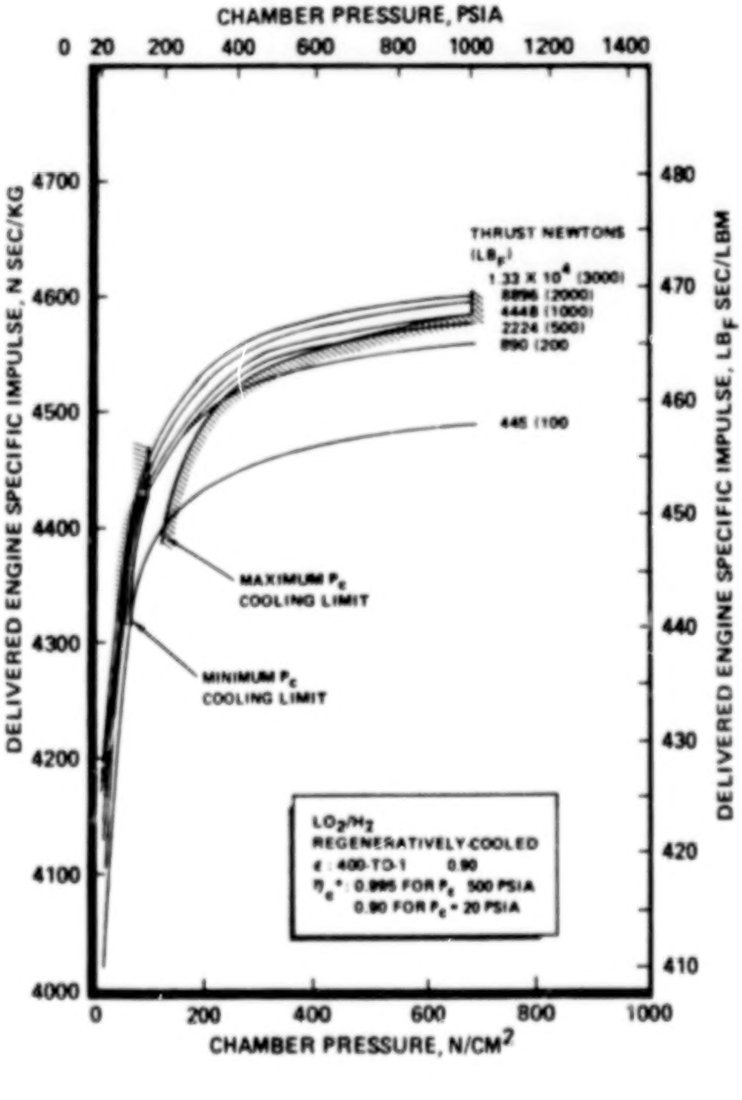
COOLING AND CYCLE LIMITS SUPERIMPOSED



Delivered engine specific impulse curves for the regenerative-cooled LO_2/H_2 gas generator cycle engines are presented in this chart. The specific impulse values were approximately 1-percent lower than for the expander cycle engines.

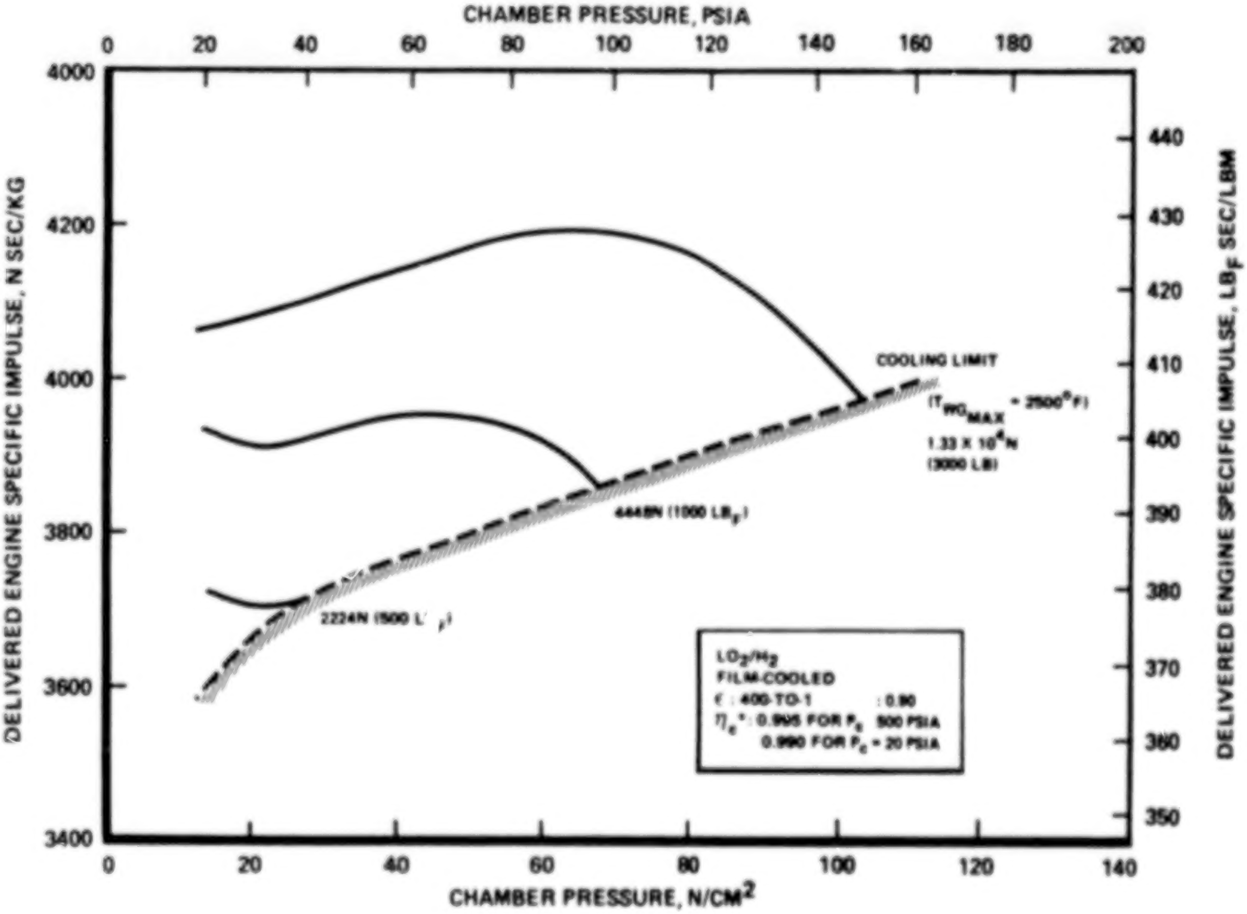
LO_2/H_2 REGENERATIVELY-COOLED GAS GENERATOR ENGINE DELIVERED SPECIFIC IMPULSE

COOLING AND CYCLE LIMITS SUPERIMPOSED



The delivered engine specific impulse for film/radiation-cooled LO_2/H_2 engines is shown in this chart. The specific impulse initially increased with chamber pressure but as the wall temperatures increased, additional film coolant was required which decreased the specific impulse with increase in chamber pressure until the maximum allowable film-cooling performance loss of 10-percent is reached (cooling limit). The maximum delivered specific impulse is approximately 428 LB_f sec/LB_m which is significantly lower than that for the regenerative-cooled engines.

LO_2/H_2 FILM-COOLED DELIVERED ENGINE SPECIFIC IMPULSE

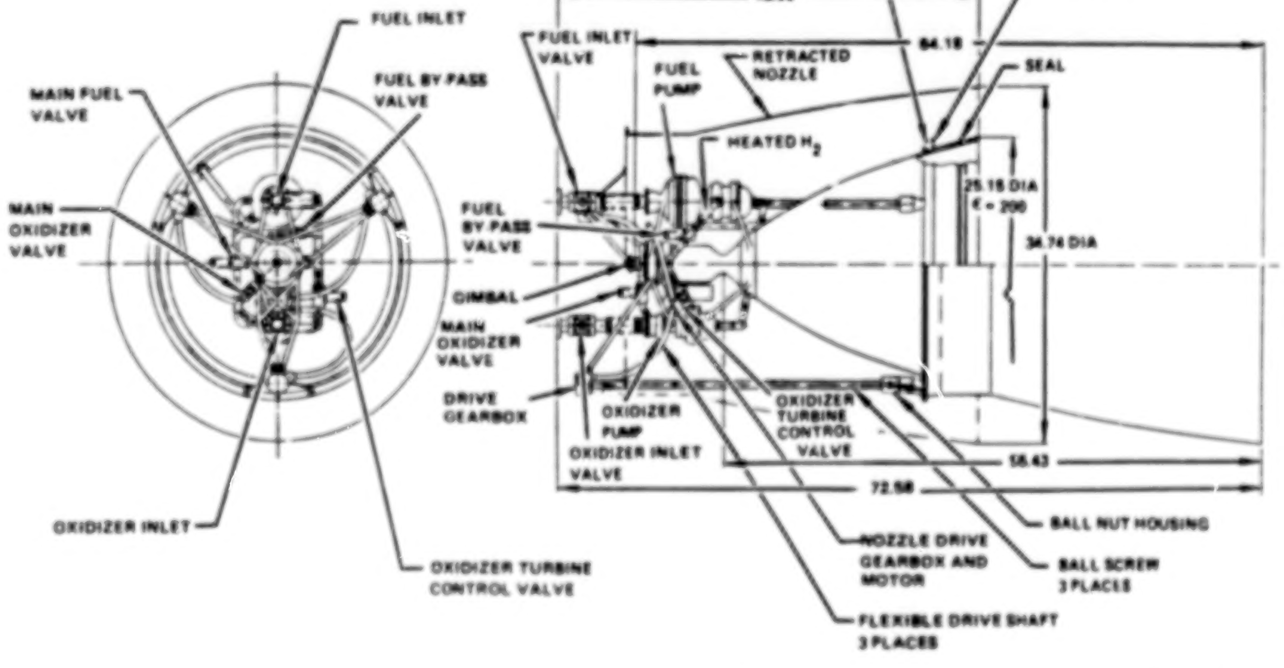


Typically one might expect that low thrust engines are all small in size. As shown in this chart, the engine length can vary from 16 inches to 340 inches. A typical LO_2/H_2 expander cycle engine at 3000-LB thrust and 660 psia chamber pressure is illustrated. The engine length is 72.6 inches and the utilization of a retractable nozzle resulted in a 42.8 inch length (a 41-percent length reduction). Since the launch vehicle is most likely the Space Shuttle, engine length can be extremely important.

EXPANDER CYCLE LOW THRUST ENGINE

801.2

PROPELLANT	LO_2/H_2
THRUST, NEWTONS (POUNDS)	1.33×10^4 (3000)
CHAMBER PRESSURE, N/CM^2 (PSIA)	455 (660)
	400


FUEL INLET
MAIN FUEL VALVE
FUEL BY PASS VALVE
MAIN OXIDIZER VALVE
OXIDIZER INLET
OXIDIZER TURBINE CONTROL VALVE
CHAMBER COOLANT INLET MANIFOLD
NOZZLE LATCH TYP
42.80
FUEL INLET VALVE
64.18
FUEL PUMP
RETRACTED NOZZLE
SEAL
HEATED H_2
25.16 DIA
ε = 200
34.74 DIA
FUEL BY-PASS VALVE
GIMBAL
MAIN OXIDIZER VALVE
DRIVE GEARBOX
OXIDIZER PUMP
OXIDIZER INLET VALVE
OXIDIZER TURBINE CONTROL VALVE
58.43
72.58
BALL NUT HOUSING
NOZZLE DRIVE GEARBOX AND MOTOR
BALL SCREW 3 PLACES
FLEXIBLE DRIVE SHAFT 3 PLACES

RANGE OF ENGINE DIMENSIONS FOR F & P_c RANGE:
DIAMETER, CM (IN.): 12.98 (5.11) TO 516.4 (201.34)
LENGTH, CM (IN.): 41.35 (16.28) TO 863.65 (340.02)

The summary of results to date are presented in this chart. From the thrust chamber cooling analyses, regenerative/radiation-cooled LO_2/H_2 thrust chambers offerred the largest thrust and chamber pressure operational envelope primarily due to the superior cooling capability of hydrogen and its low critical pressure. Regenerative/radiation-cooled LO_2/CH_4 offerred the next largest operational envelope. LO_2/RP-1 regenerative-cooling was found not to be feasible over the study range due to RP-1 coking. The inclusion of the carbon layer benefit would make LO_2/RP-1 cooling feasible; this is currently being evaluated. The maximum chamber pressure for film/radiation-cooling was significantly lower than for regenerative/radiation-cooling. As in regenerative/radiation-cooling, LO_2/H_2 thrust chambers achieved the highest maximum chamber pressure. LO_2/CH_4 film/radiation-cooling was found not feasible and LO_2/RP-1 film/radiation-cooling was extremely limited.

In the engine cycle/configuration evaluation, the engine cycle matrix was defined through the incorporation of the heat transfer results. Engine cycle limits were established with the fuel-cell power cycle achieving the highest chamber pressure; however, the fuel cell system weights were excessive. The staged combustion cycle achieved the next highest chamber pressure but the preburner operational feasibility was in question. The next highest chamber pressure was achieved by the direct drive expander cycle.

Currently in addition to finalizing the cycle limits, the complexity and weight of the engine cycles are currently being determined. This engine cycle/configuration evaluation is to lead to the selection of one LO_2/H_2 and one LO_2/hydrocarbon fuel engine for preliminary design and analysis.

SUMMARY OF RESULTS TO DATE

- HEAT TRANSFER
  - REGENERATIVE/RADIATION COOLING
    - LO_2/H_2 OFFERED LARGEST F AND P_c OPERATIONAL ENVELOPE
      - H_2 COOLING CAPABILITY
      - LOW H_2 CRITICAL PRESSURE
    - LO_2/RP-1
      - NOT FEASIBLE OVER STUDY F AND P_c RANGE DUE TO RP-1 COKING LIMIT
  - FILM/RADIATION COOLING
    - MAXIMUM P_c LOWER THAN REGENERATIVE/RADIATION COOLING
    - LO_2/H_2: ACHIEVED HIGHEST MAXIMUM P_c
    - LO_2/CH_4: NOT FEASIBLE OVER STUDY RANGE
    - LO_2/RP-1: LOW P_c
- ENGINE CONFIGURATION EVALUATION
  - DEFINED ENGINE CYCLE/CONFIGURATION MATRIX
    - INCORPORATED HEAT TRANSFER RESULTS
  - ENGINE CYCLE/CONFIGURATION LIMIT (ORDER OF HIGHEST P_c TO LOWEST AT A GIVEN THRUST)
    - FUEL-CELL POWERED CYCLE
    - STAGED COMBUSTION CYCLE (FOR LO_2/H_2)
    - DIRECT DRIVE EXPANDER CYCLE
  - FUEL-CELL RESULTED IN EXCESSIVE WEIGHT
  - STAGED COMBUSTION PREBURNER DESIGN FEASIBILITY BEING EVALUATED
  - ENGINE CYCLE/CONFIGURATION COMPLEXITY ANALYSIS IN PROGRESS

LOW-THRUST CHEMICAL ORBIT TO ORBIT PROPULSION SYSTEM

PROPELLANT MANAGEMENT STUDY

Ralph H. Dergance
Martin Marietta Corporation

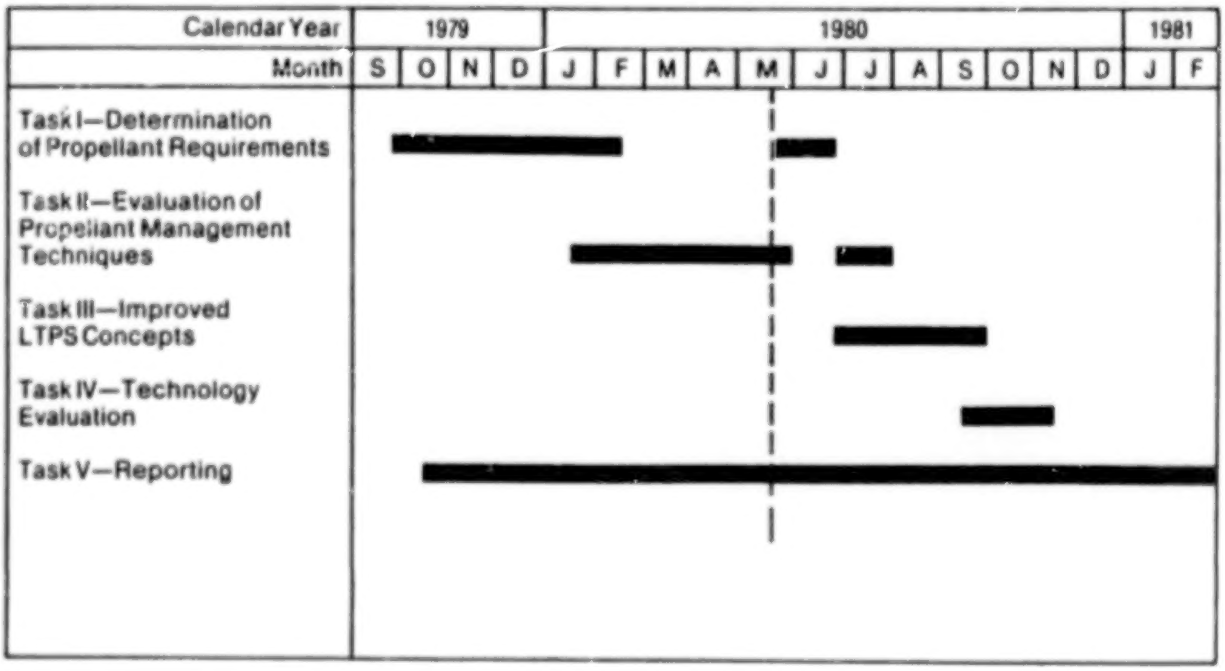
Contract Information

- Customer: NASA-Lewis Research Center
  Mr. John C. Aydelott

- Contract Number: NAS3-21954

- Period of Performance: 14 September 1979—14 November 1980

Program Schedule



LTPS Summary

The primary objective of the Low Thrust Chemical Orbit to Orbit Propulsion System Propellant Management Study Program is to determine propellant requirements, tankage configurations, preferred propellant management techniques, propulsion systems weights, and technology deficiencies for low-thrust expendable propulsion systems.

LTPS Task Objectives

Task I—Determination of Propellant Requirements—Determine propellant subsystem mass and volume for three propellant combinations and two insulation systems that minimize potential stage length.

Task II—Evaluation of Propellant Management Techniques—Determine feasibility of potential propellant management techniques and attendant weight penalties for tankage configurations determined in Task I.

Task III—Improved LTPS Concepts—Determine the maximum performance (minimum mass) LTPS for the three propellant combinations. Further refine Task I analyses.

Task IV—Technology Evaluation—Determine adequacy or deficiencies associated with the concepts defined in Task II and III.

Task V—Reporting—Monthly technical and financial reports, work plan, and final report.

Task I—Determination of Propellant Requirements

Ground Rules

Performance Specifications—MR, I_{sp}, total ΔV, and LEO to GEO transfer time supplied by NASA-LeRC; 60,000 lbm liftoff weight for propulsion system and payload.

Mission Timeline—Propellant topping is allowed to T-4 min; tanks locked up until T+90 sec; tank ΔP not to exceed 6 psid; 40-hr erection time; LEO to GEO transfer time specified by NASA-LeRC.

Design Criteria—Minimum length of propulsion system.

54 Study Candidates

3 Propellant Combinations	LO_2/LH_2, LO_2/LCH_4, $LO_2/RP\text{-}1$	All Combinations
3 Thrust Levels	100, 500, 1000 lbf	
3 Burn Strategies	1, 4, 8 Perigee Burns	
2 Insulation Concepts	MLI and SOFI	

Selected LTPS Point Design Parameters Supplied by NASA LeRC

Propellant Combination	Thrust (Lbs)	No. of Burns	ISP (Sec) 400:1	Total ΔV Required (ft/sec)	LEO to GEO Transfer Time (Hours)
LOX/LH_2 MR=6:1	100	1	422.5	18,166.3	59.21
		4		17,294.8	61.38
		8		16,349.9	72.37
	500	1	440.0	17,352.4	16.89
		4		15,931.2	19.83
		8		14,593.9	31.76
	1000	1	449.0	16,892.4	11.74
		4		15,526.1	14.91
		8		14,479.7	27.11
LOX/CH_4 MR=3.7:1	100	1	337.5	18,126.3	52.85
		4		17,262.8	55.37
		8		16.326.6	66.74
	500	1	356.5	17,258.6	15.77
		4		15,874.2	18.83
		8		14,571.4	30.87
	1000	1	364.5	16,759.0	11.19
		4		15,450.4	14.41
		8		14,448.1	26.67
LOX/RP-1 MR=3:1	100	1	317.5	18,115.5	51.08
		4		17,254.1	53.69
		8		16,320.3	65.16
	500	1	333.5	17,228.5	15.40
		4		15,855.8	18.50
		8		14,564.2	30.79
	1000	1	343.0	16.720.9	11.03
		4		15,428.8	14.27
		8		14,438.9	26.53

Initial Screening of Tank Configurations

Objective—Find Minimum Length Tanking System

Method

- Compute Required Volume
  - —Compute Usable ΔV Propellant
  - —Assume 14-ft Diameter - 2% Ullage
  - —Assume Boiloff Is 5% of ΔV Propellant
- Compute Tank Sizes

Configurations

- Maximum and Minimum Propellant Requirements (1000 lbf, 100 lbf) Were Computed for Three Propellant Combinations:
  - —LO_2/LH_2
  - —LO_2/LCH_4
  - —LO_2/RP-1
- Three Tanking Configurations Were Sized for Each Propellant Combination

Results

- Minimum Length Systems Were Elliptical Domed/Toroidal for All Propellant Combinations
- Maximum Length Systems Were for LO_2/LH_2 Parallel Tanks; LO_2/LCH_4, LO_2/RP-1 Elliptical Tanks.

Preliminary Tanking Configurations

In preparation for the Propulsion System Characterization studies, some preliminary configuration sizing calculations were performed. Based on previous Tug Studies* several of the more promising configurations were considered for each of the LTPS propellant combinations and for both maximum and minimum propellant loads. The usable propellant quantities were calculated using the ideal velocity equation and the velocity increments and specific impulses for each propellant combination, burn strategy and thrust level. The minimum loads were derived from the maximum thrust, maximum I_{sp} and 8 perigee burn conditions; while the maximum loads were derived from the minimum thrust, minimum I_{sp} and 1 perigee burn conditions.

The series "conventional" tankage configuration utilizes either ellipsoidal ($\sqrt{2}$) or cylindrical/ellipsoidal ($\sqrt{2}$) tanks up to a maximum diameter of 14 feet. The parallel tank configuration utilizes four cylindrical/ellipsoidal ($\sqrt{2}$) tanks packaged within a 14-foot outer diameter. The specific oxidizer and fuel tank diameters were selected to minimize the overall stage length. A distance of 0.5 feet was used between adjoining tanks to allow for insulation and clearance. The series "non-conventional" tankage configuration utilizing a toroidal tank and either an ellipsoidal ($\sqrt{2}$) or a cylindrical/ellipsoidal ($\sqrt{2}$) tank was determined to be the minimum length configurations for all propellant combinations.

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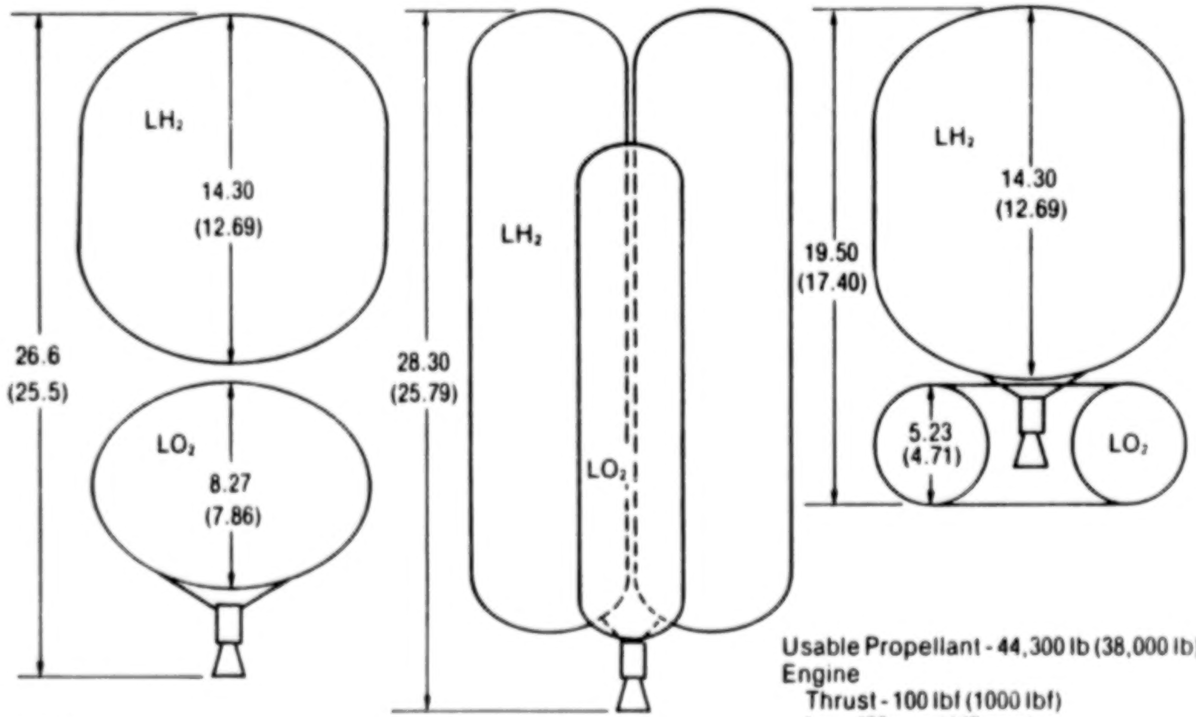
*"Space Tug Systems Study (Storable)", MCR-73-235, Final Report of Work Performed by Martin Marietta Corp. for Marshall Space Flight Center under Contract NAS8-29675, Sept. 1973.

Preliminary Tankage Configuration— LO_2/LH_2



Note:
Dimensions in feet.
Maximum diameter = 14 ft.
Engine length 3 ft (4 ft).

Usable Propellant - 44,300 lb (38,000 lb)
Engine
Thrust - 100 lbf (1000 lbf)
I_{sp} - 423 sec (449 sec)
MR - 6.0
Perigee Burns - 1 (8)
MLI Insulation

Embedded Engine Analysis

To imbed the engine in the center space of the parallel tank arrangement, the individual tank diameters must be reduced to create a space for at least the engine thrust chamber assembly. To determine the corresponding increase in length of the tank requires calculating the volume as a function of the length. By combining the volume relationships for $\sqrt{2}$ domes and right circular cylinders, the following expression was derived:

$$L_T = \frac{V_T}{\pi r^2} + \frac{2}{3\sqrt{2}} r = \frac{V}{\pi r^2} + 0.4714r$$

where:

L_T = tank length
V_T = tank volume
r = tank radius

or

$$\frac{dL_T}{dr} = -\frac{2V}{\pi r^3} + 0.4714$$

The value of dL_T/dr is large and increases rapidly as the diameter of the tank decreases.

The facing page presents the results of this analysis for the cases shown. In all instances, the stage length is increased by imbedding the engine.

Embedded Engine Analysis

Objective

Reduce parallel tank diameter (cylindrical with $\sqrt{2}$ domes) to accommodate embedded engines in an attempt to reduce length.

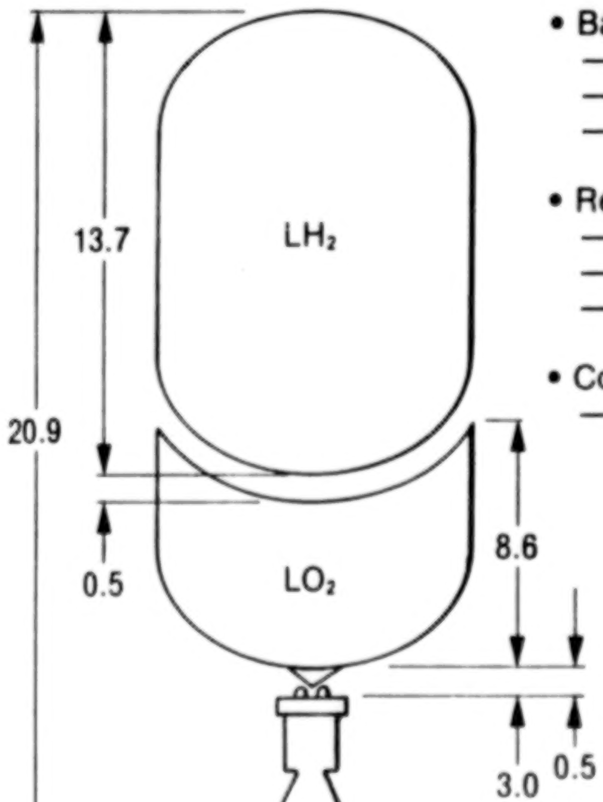
Propellant Combination	Thrust Level, lbf	Propellant Mass, lb	Δ Tank Length, ft	Engine Length, ft	Δ Stage Length, ft
LO_2/LCH_4	100	48,700	4.2	3.0	+1.2
LO_2/LCH_4	1000	42,500	4.7	4.0	+0.7
LO_2/RP-1	100	49,800	4.1	3.0	+0.9
LO_2/RP-1	1000	42,500	4.7	4.0	+0.7
LO_2/LH_2	100	46,100	6.6	3.0	+3.6
LO_2/LH_2	1000	38,000	7.1	4.0	+3.1

Conclusion—Elliptical/Toroid Tankage Scheme is Shorter

Concentric Bulkhead Configuration

For this analysis, one tank containing conventional $\sqrt{2}$ domes and the other with an inverted $\sqrt{2}$ dome were used. The overall stage length was calculated using (a) an inverted dome tank for the oxidizer tank with no change to the fuel tank, and (b) an inverted dome fuel tank with no change to the oxidizer tank. The shortest configuration was still 1.4 Ft. longer than the tandem/toroidal arrangement.

Concentric Bulkhead Configuration



Dimensions in Feet

- Baseline
  - —LO$_2$/LH$_2$
  - —1000 lbf
  - —1 Perigee Burn
- Results
  - —Length = 20.9 ft
  - —Tandem / Toroidal Length = 19.5 ft
  - —Reversed Bulkhead Length = 22.6 ft
- Conclusion
  - —Tandem / Toroidal Configuration Is Shorter

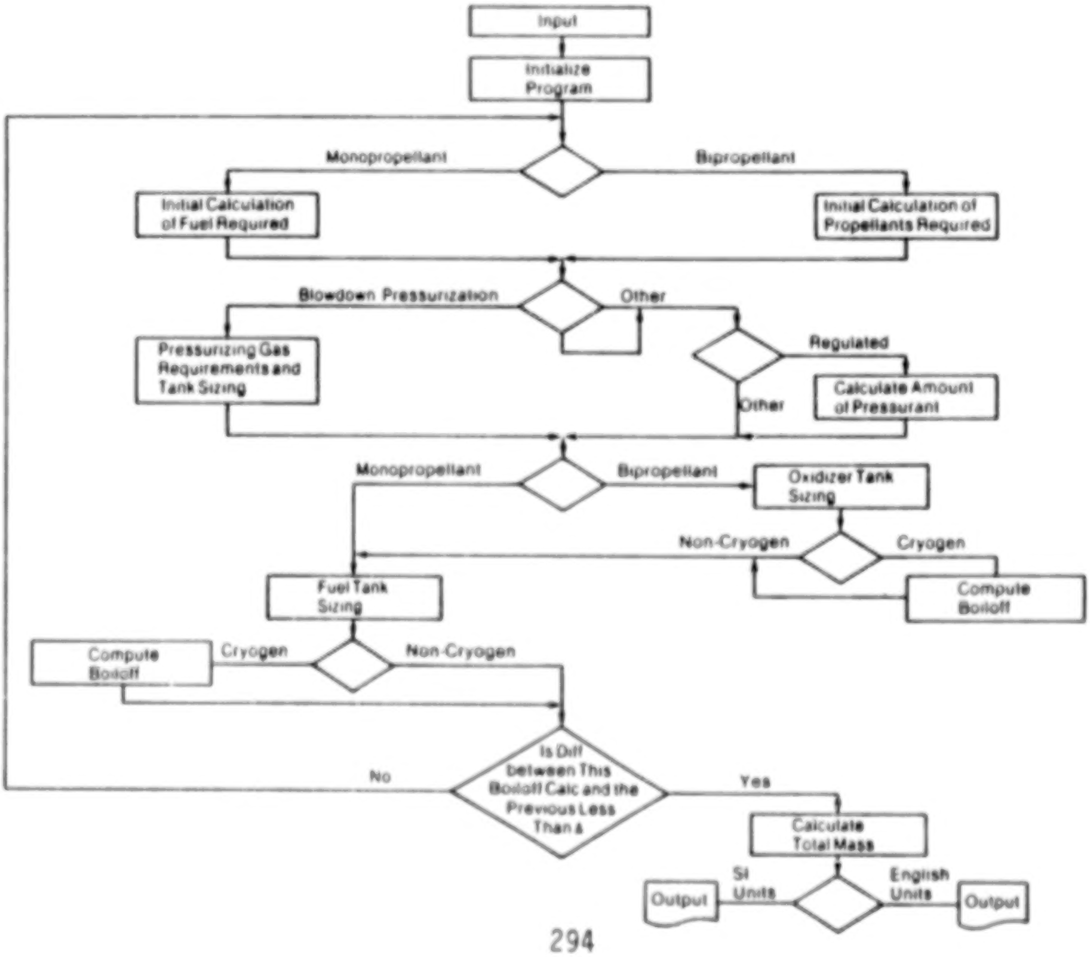
PROP Program Summary Chart

This program (PROP) was written and checked out during the early Viking Program and has been used many times since as a design and analysis tool. The program has four major system options: first, the choice of a monopropellant or bipropellant propulsion system using cryogenic and/or earth storable propellants. Second, the pressurization system sizing includes either a blowdown or a regulated case; in addition a third option bypasses the pressurization sizing loop and substitutes a fixed input mass to accommodate other types of systems (autogenous, etc). Third, available propellant tank shapes are: 1) spherical, 2) cylindrical with hemispherical ends, 3) cylindrical with $\sqrt{2}$ ellipsoidal ends, 4) $\sqrt{2}$ ellipsoidal tank, and 5) toroidal. The fourth option allows the input/output units to be specified in one of four combinations, 1) English/English, 2) English/SI, 3) English/English and SI, and 4) SI/SI. Other options are chosen at input, such as the specific vehicle mass, delta-V, and ISP and allowing the computer to calculate the propellant mass; or specifying the mass of propellant burned. Also, the program will model a wide range of adiabatic or isothermal burns.

The program output includes a complete propellant inventory (including boil-off for cryogenic cases), pressurant and propellant tank dimensions for a given ullage, pressurant requirements, insulation requirements and miscellaneous masses. The output also includes the masses of all tanks; the mass of the insulation, engines and other components; total wet system and burnout mass; system mass fraction; total impulse and burn time.

In addition, a modification was programmed to provide the capability to calculate the remaining mass, volume, and ullage height at the beginning of all burns for each propellant. The ullage height is the length of the inside of the tank minus the height of the propellant if it were all settled in the bottom of the tank. Also calculated at the initiation of each burn is the total system mass and acceleration along with the burn duration. The same variables, except ullage height and burn duration, are also computed at the end of the circularization burn. The final outputs are propellant tank dimensions.

PROP Program Summary Flow Chart


Input
Initialize Program
Monopropellant
Bipropellant
Initial Calculation of Fuel Required
Initial Calculation of Propellants Required
Blowdown Pressurization
Other
Pressurizing Gas Requirements and Tank Sizing
Regulated
Other
Calculate Amount of Pressurant
Monopropellant
Bipropellant
Oxidizer Tank Sizing
Non-Cryogen
Cryogen
Compute Boiloff
Fuel Tank Sizing
Compute Boiloff
Cryogen
Non-Cryogen
No
Is Diff between This Boiloff Calc and the Previous Less Than ε
Yes
Calculate Total Mass
SI Units
English Units
Output
Output

Baseline Insulation Characteristics

A number of different insulation sytems were considered as LTPS candidates. The two most promising concepts appear to be a multilayer mylar system with a helium purge bag and the spray on foam insulation (SOFI) utilized on the Space Shuttle External Tank program. The SOFI (CPR-488) was compared with other foam insulations* and was selected because it had the best balance between low density and good thermal conductivity.

Multilayer insulation results in a relatively heavy system with adequate ground thermal conductivity but excellent on-orbit thermal conductivity. Thus, longer duration missions (i.e., multiple burn options which minimize ΔV but require longer transit times) stand to benefit the most from a multilayer system. The actual insulation system weight is a function of the required insulation thickness and average density; however, the optimum thickness is determined for some cases by a trade-off between boil-off/vent losses and insulation weight, and for other cases by the pressure rise during the ground hold and ascent period. The optimum insulation thickness for each of the 54 propulsion systems was determined using a analytical model programmed on a desk calculator.

Data for MLI was from; MCR-79-594 "Cryogenic Fluid Management Experiment, Thermal Analysis Report." June 1979. Martin Marietta Corp., Denver Division, Denver, Colo 80201.

SOFI Data was from; MMC Dwg. No. 82600200102 "Thermal Data Book, External Tank Project." October 1979. Michoud Operations, Martin Marietta Corp., Denver Division, Denver, Colo 80201.

*Sharpe, Ellsworth L., Helenbrook, Robert G.: "Cryogenic Foam Insulation for LH_2 Fueled Subsonic Transports", Delivered at International Cryogenic Materials Conference, July 10-11, 1978.

Baseline Insulation Characteristics

Type / Parameter	Multilayer (MLI)		Spray-on Foam Insulation (CPR-488)
	Ground	On-Orbit	
Conductivity, Btu/hr-ft²-°R	0.35	$1.8824T^{0.6} \times 10^{-6}$	$(1.7 + 0.02452T) \times 10^{-3}$
Density, lb/ft³	3.51*	3.51*	2.2†

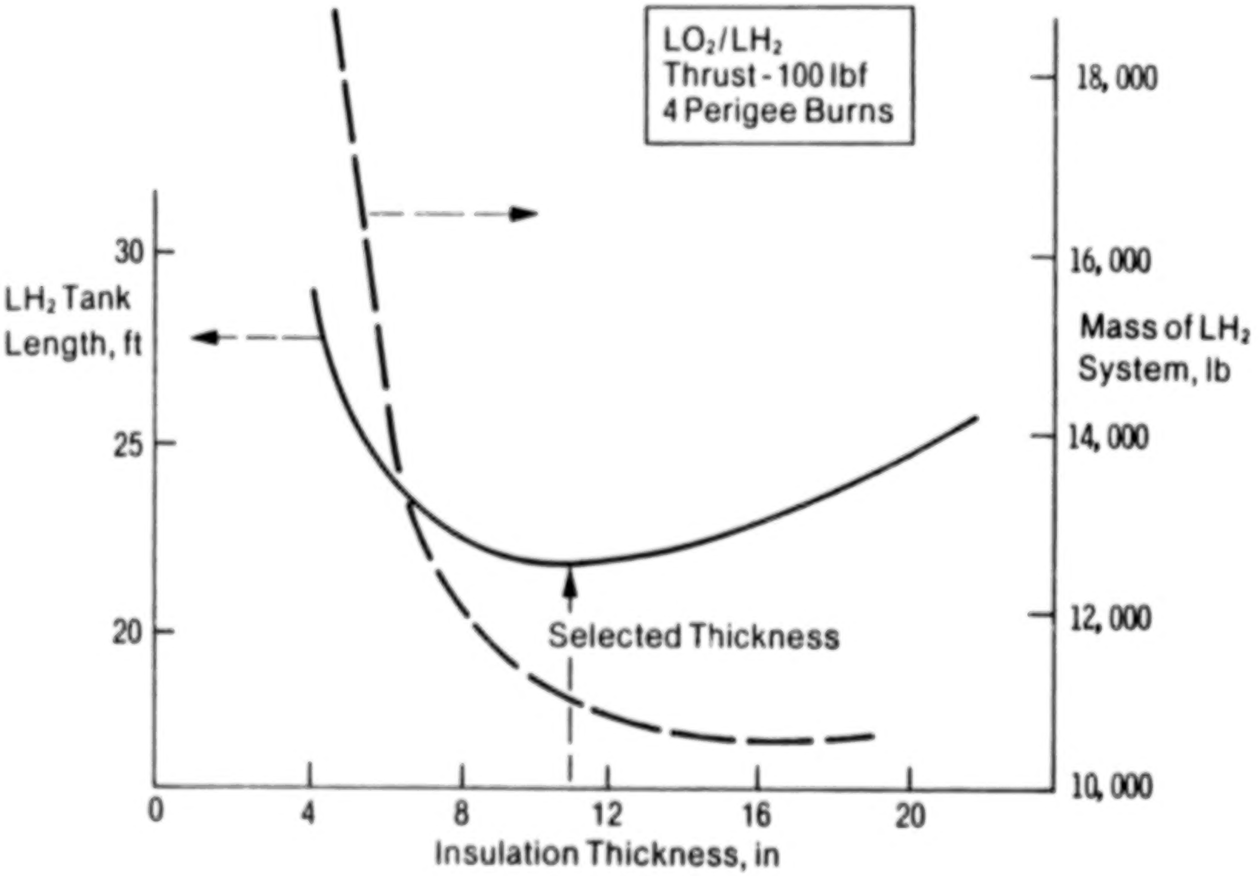
*Does not include protective cover sheet or fastening material.

†Values at 289°K (520°R).

Length - Optimized SOFI Insulation Thickness for LH_2 Tank

The two plots show Length vs Insulation Thickness (solid line) and Mass vs Thickness (broken line) for one particular mission. The mass-optimized thickness can be seen to occur at about 17 inches and the length optimized thickness is at about 11 inches. The large value of the slopes of the plots to the left of the optima are due to increasing boiloff. To the right of the optimum the slope is smaller and soon becomes constant due to additional insulation mass which is basically a linear function of thickness. As the insulation thickness decreases from 17 inches to 11 inches the length decreases about 20 inches and the mass increases approximately 500 lb_m. This means that for the LH_2 tank a substantial gain in length is accomplished without too large a weight penalty. Similar results were obtained for other SOFI-covered tanks, but where not as pronounced. Thus, when SOFI was used a length-optimized insulation thickness was also used. The selected thickness shown on the graph is the thickness predicted by the length-optimized analysis.

Length—Optimized SOFI Insulation Thickness for LH2 Tank



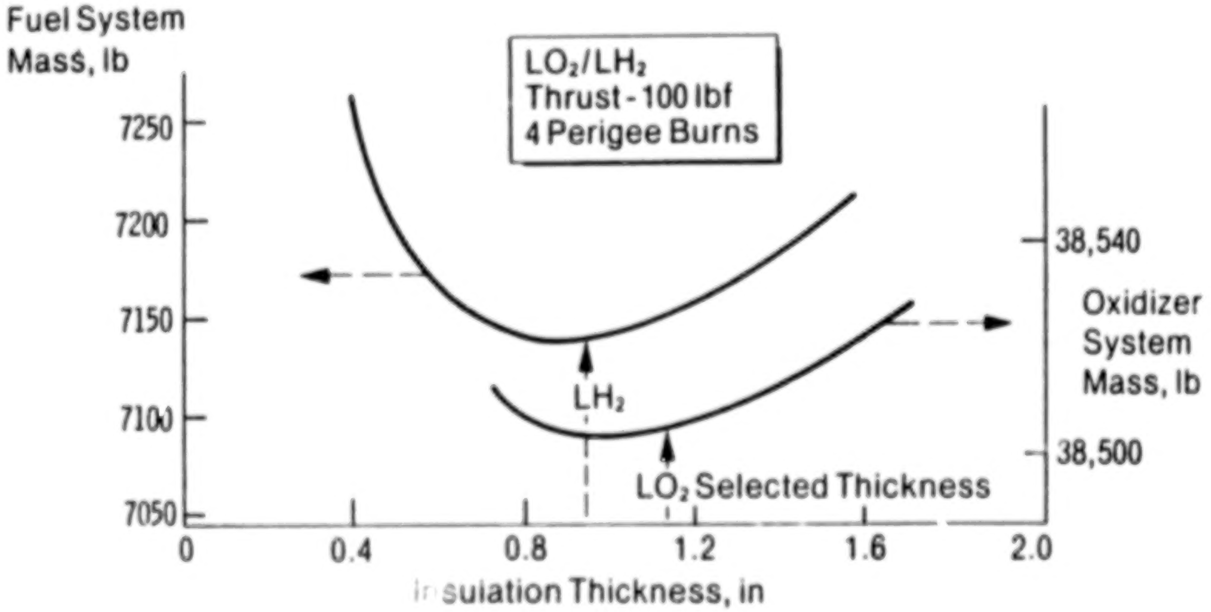
Liftoff Mass Optimized MLI Insulation Thickness

A mass optimized analytical model to predict optimum thermal insulation thickness was developed and programmed on a desk-top calculator. The final result is a single equation that calculates the insulation system thickness that results in the lowest propulsion system mass (including vent losses) for the given insulation system properties and ground and on-orbit conditions. Since a number of simplifying assumptions were required in the derivation of this equation, it was necessary to verify the relationship using the PROP computer program. The results of this checkout are shown in the following figure for MLI systems. These plots show the total mass of the system required to accommodate the propellants as a function of insulation thickness. The total mass includes insulation, tank, boil-off, trapped propellant, usable (ΔV) propellants, and start-shutdown losses. All heat transfer to the propellant is assumed to cause vaporization only with no sensible heating.

The baseline propellant combination of LO_2/LH_2 at a mixture ratio of 6:1 was used for all cases. The total payload mass was approximately 60,000 lbm. The fuel tank was a 14 foot diameter cylinder with $\sqrt{2}$ ellipsoidal domes. The oxidizer was contained in a $\sqrt{2}$ ellipsoidal tank with a major axis of 11.4 feet. The tank material was 2219-T87 aluminum. On-orbit time was assumed to be 101 hours. An equivalent on-orbit time of (ground plus ascent) of 5.4 minutes, based upon average insulation performance values for a typical STS ascent profile, was used for the representative mission.

The predicted optimum insulation thickness for each propellant tank (using the calculator program) is noted by the arrows on the Figure while the curves shown the actual total propellant system masses (calculated by PROP) plotted as a function of insulation thickness. Note that the calculator model predicts a consistently conservative value for the optimum thickness compared to the PROP predicted value. However, the maximum difference in mass from the optimum is 4 lb_m which amounts to .01% difference in total net system mass. This difference is far less than the mass differences for the various propellant systems considered in this study and did not influence the comparative results.

Liftoff Mass Optimized MLI Insulation Thickness



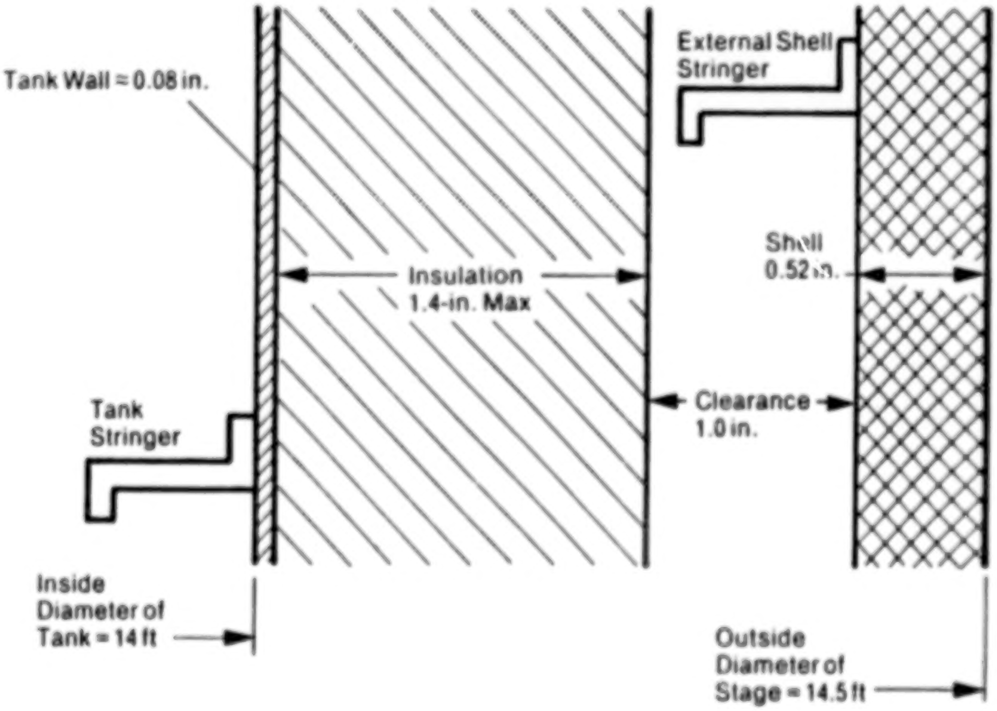
Length-optimized M epresents significant weight penalty (≈350 lb) with a small length gain (≈0.8 in).

Baseline Tank Diameter (MLI)

This chart substantiates the 14 foot tank diameter assumed for the preliminary tank screening analyses.

Starting with the maximum cargo bay diameter of 15 feet, an allowable stage diameter of 14.5 feet was determined from inputs from Martin Marietta's Payload Integration Contract. The external skin arrangement, constructed of graphite expoxy composite material, was determined from Space Tug Study results. The 1.4 inch MLI thickness resulted from the insulation studies previously discussed. By considering a typical tank wall thickness of 0.08 inches, an inside diameter of 14 feet is derived for tank sizing.

Baseline Tank Diameter (MLI)


Tank Wall ≈ 0.08 in.
External Shell Stringer
Shell 0.52 in.
Insulation 1.4-in. Max
Tank Stringer
Clearance 1.0 in.
Inside Diameter of Tank = 14 ft
Outside Diameter of Stage = 14.5 ft

Note:
For the SOFI-covered tanks, the outside diameter of the insulation is constrained to 170 in., and the inside diameter of the tank will vary depending on the insulation thickness.

Propellant Inventory

The elements of a typical propellant inventory are shown. All items are self-explanatory with the exception of expulsion efficiency and loading accuracy.

Expulsion efficiency was based upon Martin Marietta's assessment of the performance of a typical surface tension propellant management device for this application. The 98% value, although representative,will be updated based on results of analyses conducted later in the contract.

Loading accuracy was based on values that have been achieved with demonstrated loading techniques.

Propellant Inventory

- ΔV—Calculated Using the Ideal Velocity Equation
- Performance Reserve—2% of ΔV Requirement
- Start/Shutdown Losses—Scaled Down from Centaur Data
- Boiloff—Calculated as a Function of Mission Profile, Tank Structure, and Insulation
- Trapped—Estimated from Stage and Tanking Geometry
- Expulsion Efficiency—98%
- Loading Accuracy—0.5%

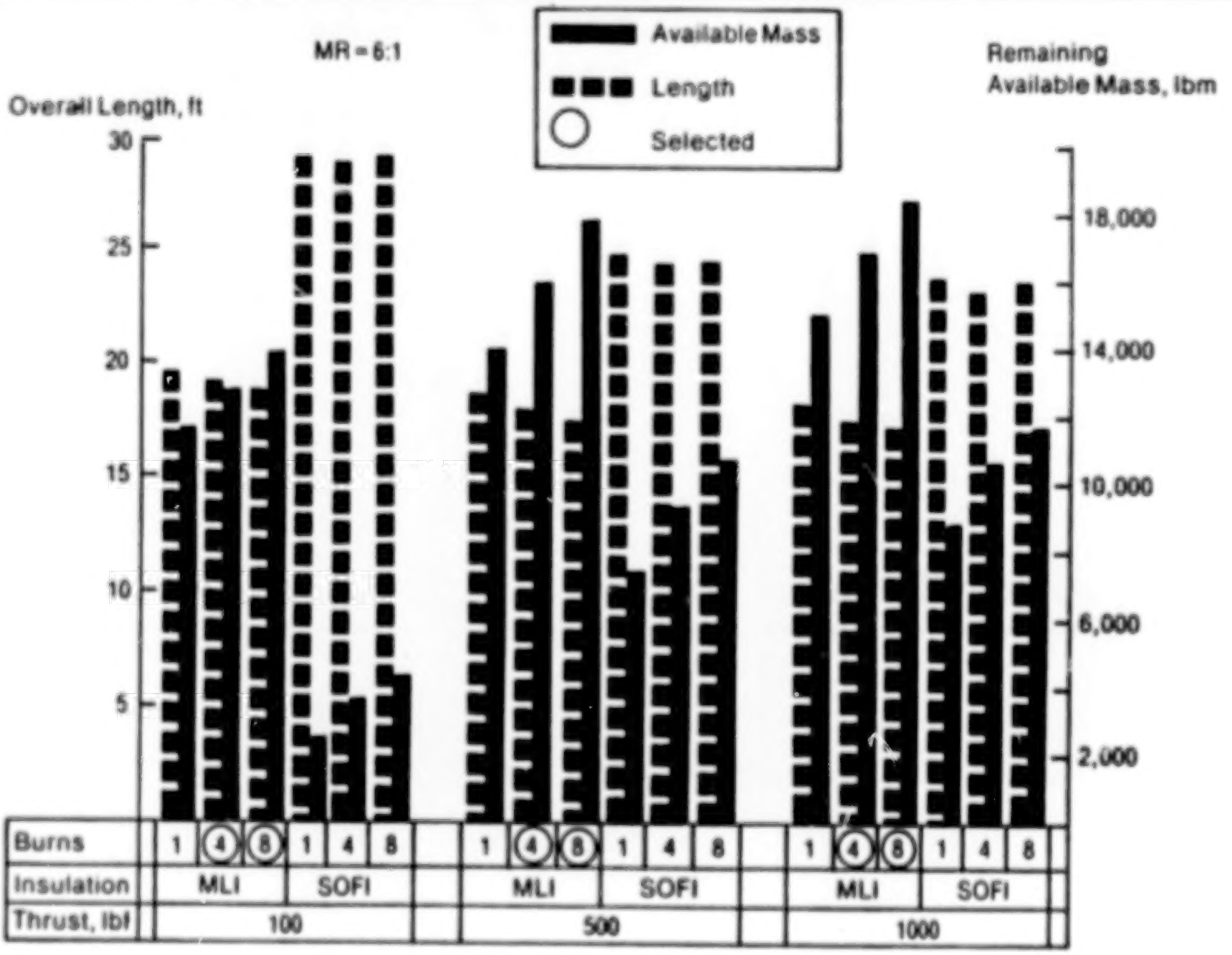
Propellant System Length & Available Mass

- Overall Length
  - All Elliptical/Toroidal Configurations
  - Tankage (Including Insulation) Only
  - Top of Toroid Coplanar with Bottom of Ellipsoid
- Remaining Available Mass
  - 65,000 lbm STS Capability
  - 5,000 lbm ASE
  - 60,000 lbm Liftoff
  - Available = 60,000 lbm—Stage Not Including Avionics, Propellant Management Device, ACPS, or Adapters.

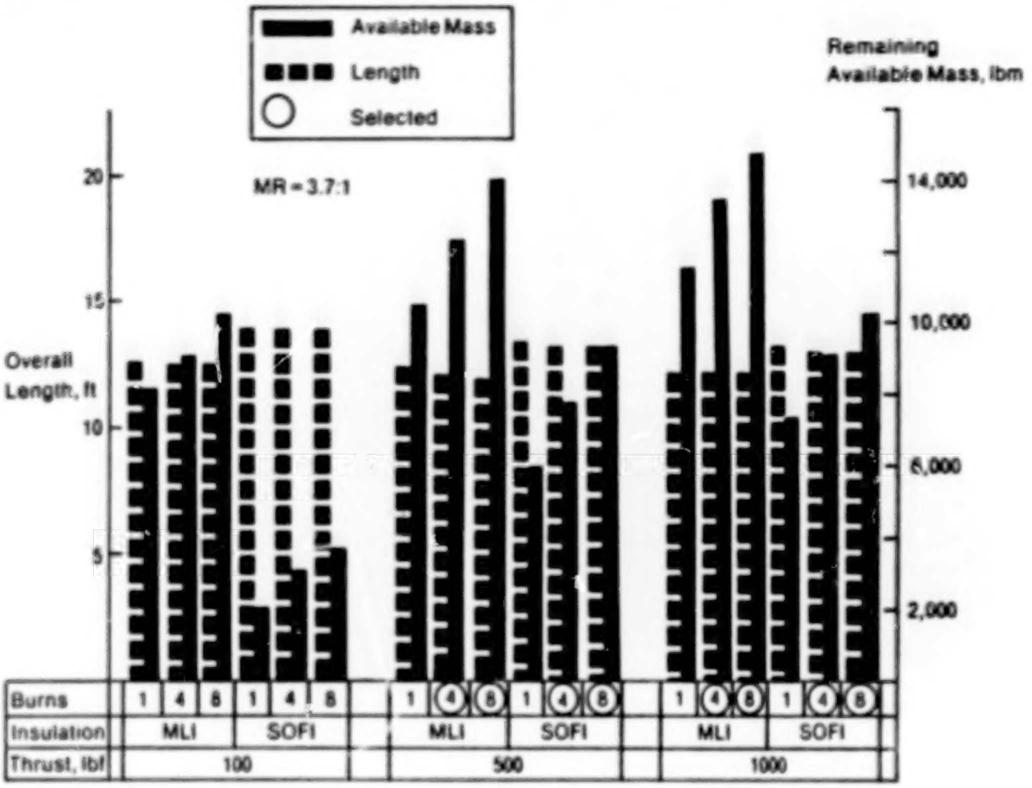
LO_2/LH_2 Propellant System Length and Available Mass

The definitions of length and available mass were presented on the previous page. The configurations circled on the next 3 charts are those selected for use in Task II - Evaluation of Propellant Management Techniques - of this program. They were selected to maximize available mass while minimizing length. However, some SOFI configurations were chosen even without satisfying the aforementioned criteria, to maintain this concept for technology evaluation.

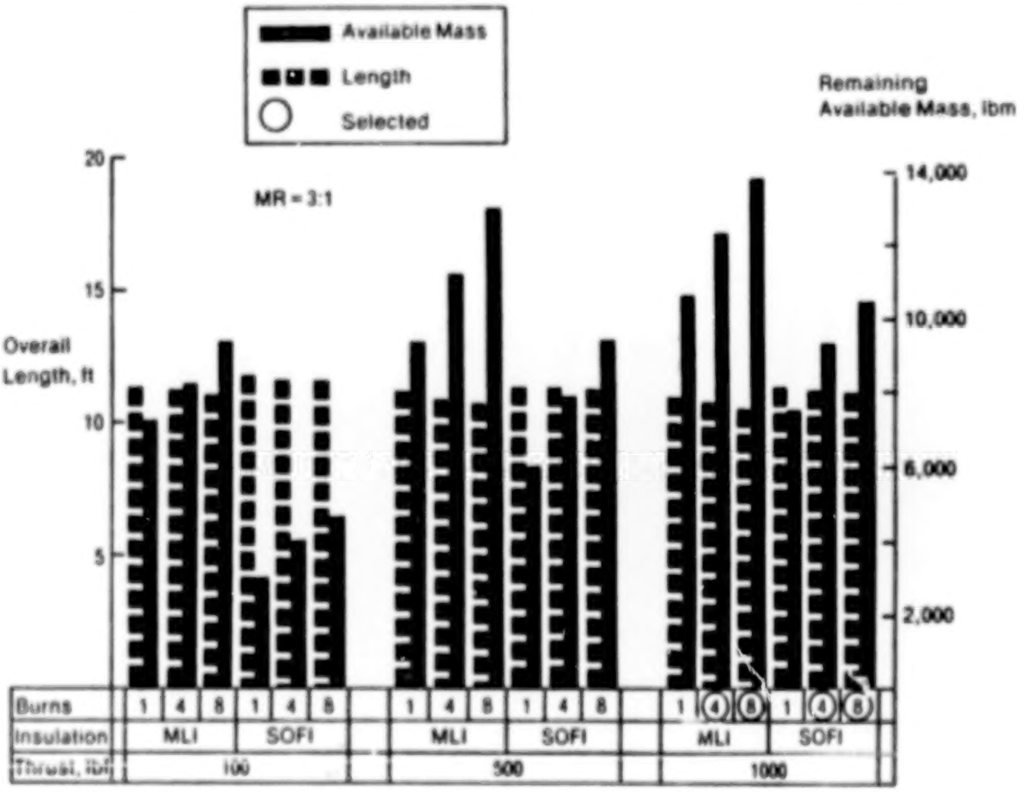
LO_2/LH_2 Propellant System Length and Available Mass



LO2/LCH4 Propellant System Length and Available Mass



LO2/RP-1 Propellant System Length and Available Mass



Task II - Evaluation of Propellant Management Techniques

Three types of propellant management methods; propulsive settling, partial acquisition devices and total acquisition devices, were applied to the selected propulsion systems. The propellant for the settling thrusters was either the primary propellants or N_2O_4 and MMH. NASA LeRC provided a computer model used to predict the propellant settling times.* The partial and total acquisition devices are fine mesh screen surface tension type propellant management devices.

For each propellant management method, its feasibility for this application was determined and the total weight penalty for each method was calculated.

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*I.E. Sumner: "Liquid Propellant Reorientation in a Low-Gravity Environment", NASA TM-78969, NASA Lewis Research Center, Cleveland, Ohio, July 1978.

Task II—Evaluation of Propellant Management Techniques

Determine feasibility and weight penalty of propellant management concepts for the selected low-thrust propulsion systems.

Concepts:

- Propulsive settling—Utilizing LeRC-supplied model
  - —Using main engine propellants for settling thrusters
  - —Using N_2O_4 and MMH as propellants for settling thrusters
- Fine mesh screen partial acquisition system
- Fine mesh screen total acquisition system

Results - Propulsive Settling

It was found that by using very small thrusters, in the range of 0.1 to 1.0 lb_f, the amount of propellant required to perform settling prior to every burn is small (less than 10 lbm). However, the residuals left in the tank due to suction dip during terminal drain can be large (200 to 800 lbm), especially in the toroidal tank. Means of reducing the draining residuals will be investigated under a subsequent task of this program. Of the three propellant management methods, propulsive settling had the highest weight penalty.

Results—Propulsive Settling

- By using very small thrusters (0.1 to 1.0 lbf), propellant requirement for settling is very small (< 10 lbm).

- Residuals due to draining can be large (200 to 800 lbm), especially in toroidal tank.

- Highest weight penalty.

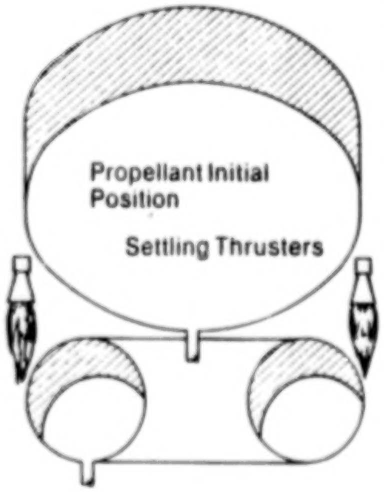
Propellant Settling Approach

Prior to each main engine burn the settling thrusters fire, producing an acceleration capable of causing reorientation of the propellant. This acceleration must be maintained for a period long enough to position the propellant at the tank outlet so that the main engines can start. To cause reorientation the acceleration must be greater than atmospheric drag, which is significant prior to the first burn in low earth orbit. In addition, the acceleration must be large enough to create interface instability in both tanks, with the smaller radius toroidal tank being the most stable. Too large an acceleration can cause liquid geysering, which will increase the time required to complete settling.

It was assumed that the settling thrusters were part of the attitude control system, and their thrust level and the number firing could be selected. Therefore, only the weight of the propellant used to perform the settling contributed to the weight penalty. The draining residuals also add to the weight penalty.

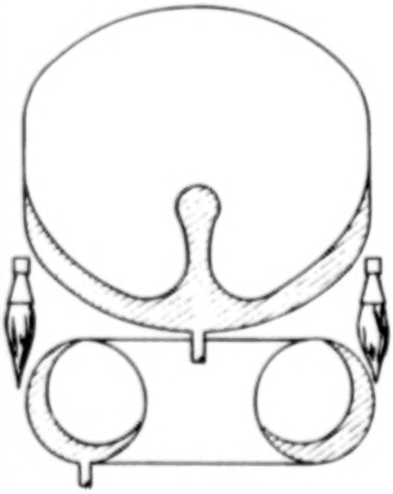
Propellant Settling Approach

Initial Conditions



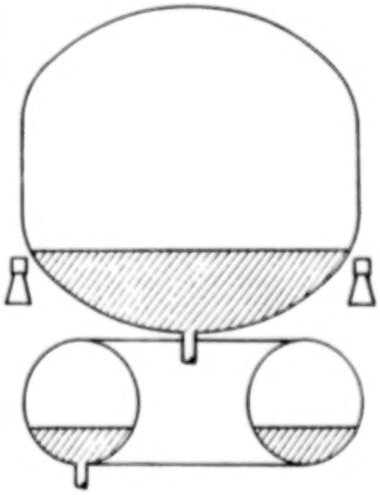
Settling Thrusters Fire
- Overcome Atmospheric Drag
- Cause Interface Instability

Settling Underway



- Settling Thrusters Continue to Fire
- Thrust Selected to Minimize Liquid Geysering
- Thrusters Assumed To Be Part of Attitude Control System

Settling Complete



- Settling Thrusters Shut off when Settling Predicted To Be Complete
- Main Engine Immediately Started

Weight Penalty:
Propellant Required to Produce Settling Draining Residuals

Results - Partial Acquisition Systems

It was found that a reservoir of reasonable size (less than 15 ft^3) will meet the expulsion requirements. Methods of refilling the reservoir during an engine burn were not feasible due to the low acceleration produced by the main engines. A significant portion of the propellant in the reservoir is lost due to vaporization. Since the sizing of the reservoir is critical to the successful operation of the device, careful accounting of all such losses is required. The reservoir will have to be constructed of a sandwich of perforated plate and screen layers so that the screen will remain wetted and retain propellant within the reservoir.

With a few exceptions, the partial acquisition devices had the lowest weight penalty of the three propellant management methods.

Results—Partial Acquisition Systems

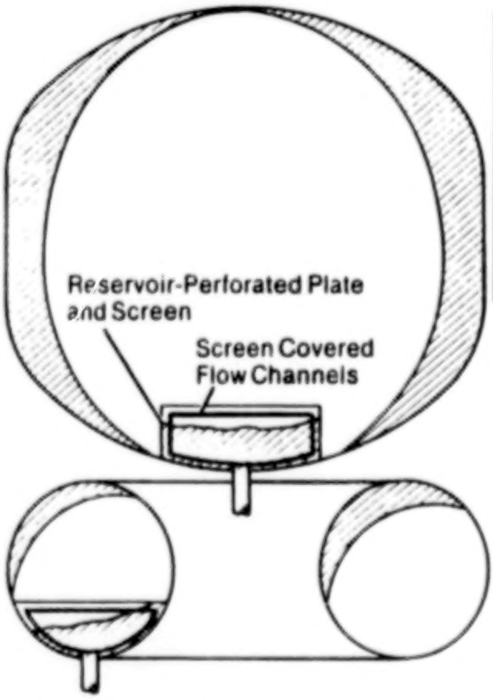
- Refillable traps not feasible for this application primarily due to low accelerations.
- Nonrefillable traps, with a relatively small volume (< 15 ft^3) will satisy requirements.
- Significant portion of propellant in trap is lost due to vaporization (typically 1/2 to 2/3).
- Sizing of trap to supply all requirements is critical.
- Dryout of reservoir screen is a concern.
- Lowest weight penalty (with a few exceptions).

Partial Acquisition Device

A partial acquisition device consists of a reservoir that holds sufficient propellant to start the main engine for each burn and a channel network within the reservoir that guarantees gas-free flow of propellant to the tank outlet. The reservoir and channels are made of a frame covered with a fine-mesh screen, which provides the necessary liquid retention characteristics. In addition to supplying propellant to the engines until the bulk propellant settles, the reservoir must also contain sufficient propellant to fill the feedline, prechill the engine and provide for losses from the reservoir due to vaporization. The weight penalty is the weight of the device plus the weight of residual propellant that cannot be expelled.

Partial Acquisition Device

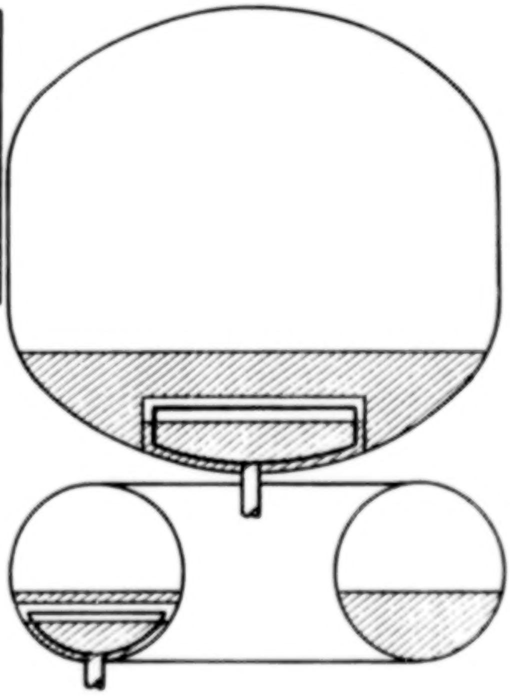
Propellant Settling


Reservoir-Perforated Plate and Screen
Screen Covered Flow Channels

- Main Engine Starts
- Device Supplies Propellant as Bulk Propellant Settles
- Gas Enters Device When It Is Not in Contact with Bulk Liquid

Trap Volume Requirement:
- Initially Fill Feedline
- Prechill Engine before Each Burn
- Settle during Each Burn
- Vaporizaton Losses

Continued Main Engine Firing



- Propellant Feed Continues
- Gas Cannot Be Purged

Weight Penalty:
  Propellant Residuals
  Device Weight

Results - Total Acquisition Systems

A simple channel network, with a small channel flow area, will meet the expulsion requirements. At terminal drain, screen area becomes critical, so screen manifolds at the outlet are necessary. The largest manifolds are required for those systems with the greatest acceleration during terminal drain.

These frail channels must be supported from the tank wall so as to withstand launch loads. Heat transfer into the channels must be limited to prevent the boiling of propellant inside the channels.

Since this device operates independent of propellant settling, it can expel propellant whenever required and, therefore, makes it more flexible than the other methods. The weight penalty for total acquisition was close to that of partial acquisition, but slightly heavier.

Results—Total Acquisition Systems

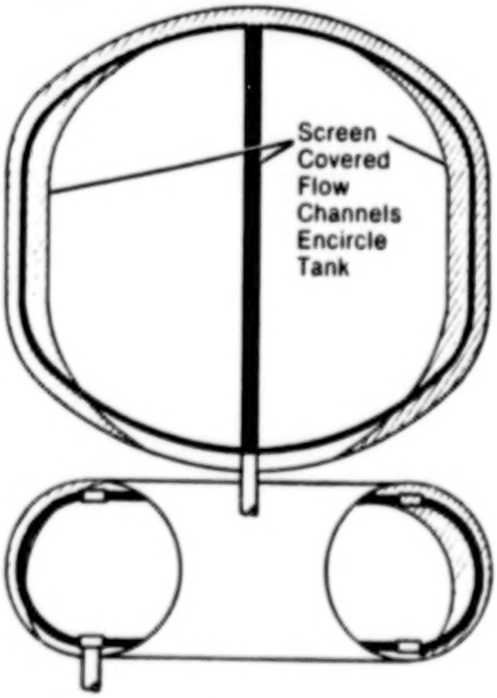
- Simple channel concept can meet requirements.
- Small channel cross-section, 4x½ in. maximum.
- Larger manifolds (10x10 in.) are required for systems with 1000 lbf thrust and SOFI, due to high accelerations during terminal drain.
- Structural support and thermal isolation of device is critical.
- Provide propellant management system flexibility.

Total Acquisition Device

A total acquisition system consists of screen covered flow channels that encircle the tank. These channels are always in contact with the bulk propellant regardless of its location so that gas-free propellant can be fed from the tank as required. The weight penalty is the weight of the device plus the weight of the propellant residuals.

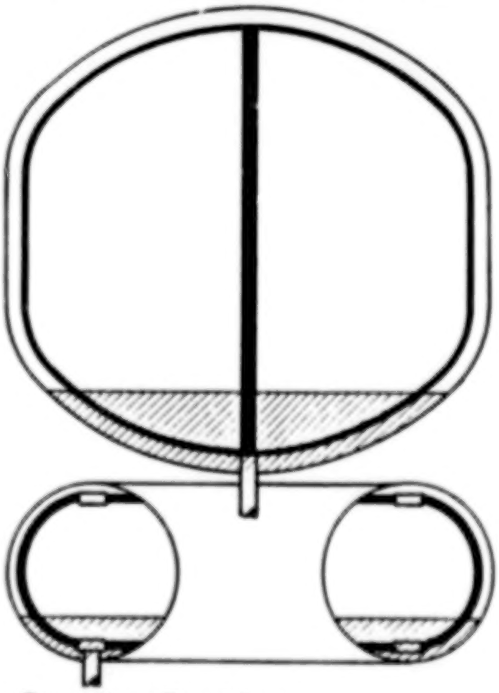
Total Acquisition Device

Propellant Settling



- Main Engine Starts
- Channels Maintain Liquid Outflow during Start and Settling

Continued Main Engine Firing



- Propellant Feed Continues
- Terminal Drain Is Worst-Case Design Condition

Weight Penalty:
Propellant Residual
Device Weight

Propellant Management Weight Penalties

The following two tables summarize the configuration of the 18 selected propulsion systems and the weight penalties of the three propellant management methods for each system.

Main engine thrust, with its resulting effect on flow rate and acceleration, had a significant effect on the draining residuals and the resulting weight penalty for propulsive settling. The weight of the total acquisition devices was also sensitive to the main engine thrust since the channel cross-section had to be increased to accommodate the greater flow rates. The variation of the weight penalty of the partial acquisition devices is rather small in comparison.

Selected Propellant System Configurations

Configuration	Propellant	Thrust, lbf	No. of Burns	Insulation System
1	LO_2/LH_2	100	4	MLI
2		100	8	
3		500	4	
4		500	8	
5		1000	4	
6		1000	8	
7	LO_2/LCH_4	500	4	MLI
8		500	8	MLI
9		500	4	SOFI
10		500	8	SOFI
11		1000	4	MLI
12		1000	8	MLI
13		1000	4	SOFI
14		1000	8	SOFI
15	LO_2/RP-1	1000	4	MLI
16		1000	8	MLI
17		1000	4	SOFI
18		1000	8	SOFI

Propellant Management Weight Penalties

Configuration	Settling		Partial Acquisition	Total Acquisition
	N_2O_4/MMH	Primary Propellant		
1	167	166	156	118
2	164	163	169	118
3	398	397	158	160
4	429	427	175	160
5	592	590	171	244
6	576	573	188	243
7	534	534	96	155
8	528	527	105	154
9	507	506	109	156
10	505	504	122	154
11	798	798	107	234
12	784	783	123	234
13	785	784	121	237
14	784	783	143	236
15	302	302	132	270
16	309	308	145	269
17	287	286	143	274
18	299	298	159	274

Weights in lbm.

SOLAR ROCKET SYSTEM CONCEPT ANALYSIS*

Jack A. Boddy
Rockwell International Corporation

The use of solar energy to heat propellant for application to earth orbital/planetary propulsion systems is of interest because of its unique performance capabilities. The achievable specific impulse values are approximately double those delivered by a chemical rocket system, and the thrust is at least an order of magnitude greater than that produced by a mercury bombardment ion propulsion thruster. The primary advantage the solar heater thruster has over a mercury ion bombardment system is that its significantly higher thrust permits a marked reduction in mission trip time.

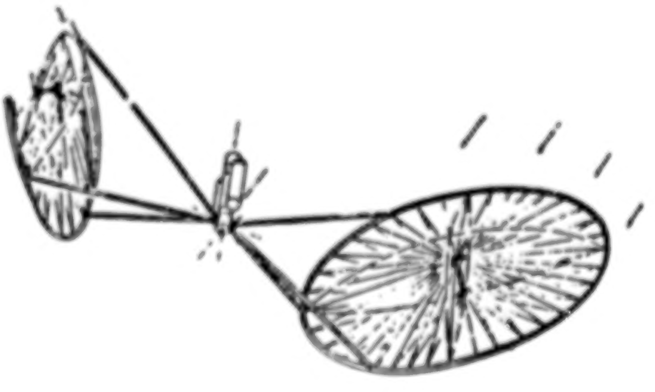
The concept of using solar energy to heat propellants for use in an earth orbital/planetary rocket propulsion system is not new. In 1962, for example, the Air Force Rocket Propulsion Laboratory (AFRPL) sponsored an analytical and experimental program to demonstrate the feasibility of the solar heated rocket engine. In a test program conducted at the AFRPL, a specific impulse of 680 seconds was achieved. The thruster utilized hydrogen as the propellant. Although the initial results were encouraging, the program was not pursued. The performance capabilities of the launch vehicles available in the early 1960's were such that the full potential of the solar rocket could not be realized. The development of the Space Transportation System (STS), however, offers the opportunity to utilize the full performance potential of the solar rocket. As the 1980-1990 time period approaches, a far greater number and variety of mission requirements have been identified than in the early 1960's that could potentially use solar rocket propulsion systems.

Objectives

The basic study objectives as stated were subjected to the guidelines of a mission model concerned with transfer from low earth orbit (LEO) to geosynchronous equatorial orbit (GEO). The return trip, GEO to LEO, both with and without payload, was also examined. Payload weights considered ranged from 2000 to 100,000 pounds. The performance of the solar rocket was compared with that provided by LO_2-LH_2, N_2O_4-MMH, and mercury ion bombardment systems.

BJECTIVES

THE OBJECTIVES OF THE SOLAR ROCKET SYSTEM CONCEPT ANALYSIS STUDY WERE TO PROVIDE AN ASSESSMENT OF THE VALUE OF SOLAR THERMAL PROPULSION RELATIVE TO MORE CONVENTIONAL PROPULSION CONCEPTS, AND TO DEVELOP AN UNDERSTANDING OF THE FACTORS WHICH BEAR ON ITS TECHNICAL FEASIBILITY.



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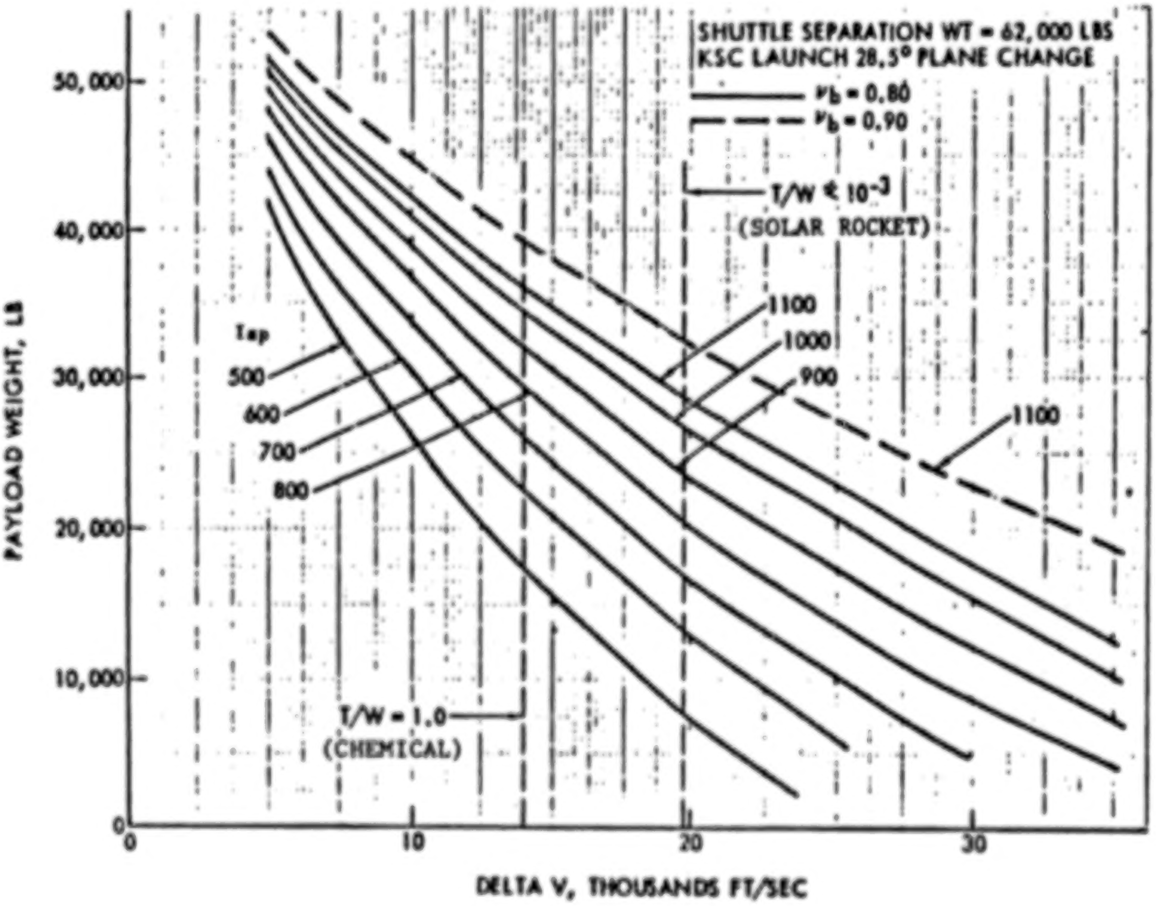
*Sponsored by the Air Force Rocket Propulsion Laboratory under Contract F04611-79-C-0007, Final Report, AFRPL-TR-79-79.

Payload Weight as a Function of ΔV and Specific Impulse

Payload weight for a range of ΔV's is presented for specific impulse values ranging from 500 to 1100 seconds, and a Shuttle separation weight of 62000 pounds. This range of specific impulses are obtainable for representative solar rocket systems. The velocity requirements for low earth orbit (LEO) to geosynchronous orbit (GEO) are about 14000 ft/sec for chemical propulsion system employing high thrust to weight ratios. For the solar rocket system with $T/W \lesssim 10^{-3}$ the velocity requirements for the continuous burn condition are 19200 ft/sec. The improvement in the higher specific impulse combined with the increase in velocity requirements still results in significant improvements in payload delivered to GEO.

PAYLOAD WEIGHT AS A FUNCTION OF ΔV AND SPECIFIC IMPULSE



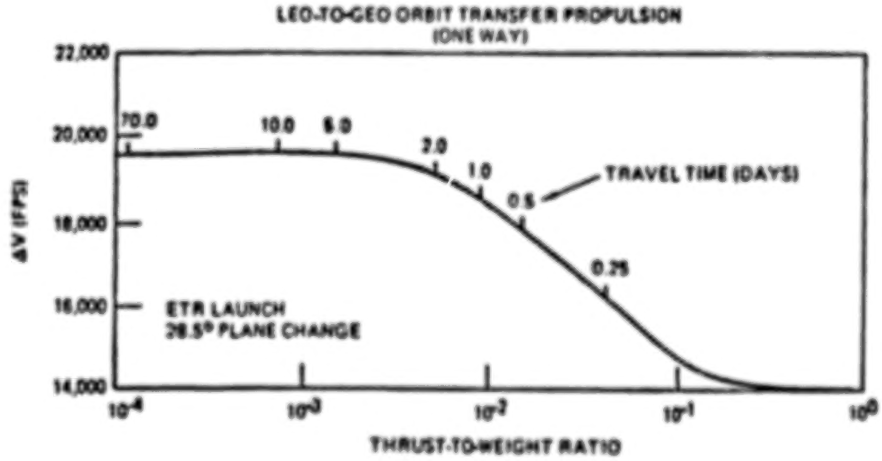
Delta - V Requirements vs Thrust-to-Weight

The classical two-impulse transfer, with one impulse at perigee and the second impulse at apogee, is commonly associated with transfer vehicles having a thrust-to-weight ratio considerably above 0.1. The mission velocity for such a vehicle corresponds to approximately 14,000 fps and a trip time of 5.27 hours.

Lower thrust-to-weight vehicles may also fall into this two-impulse transfer category as long as the corresponding burntime is generally shorter than the transfer time and the transfer trajectory still resembles an ellipse. The corresponding mission velocity would be considerably higher; and the trip time, although also increasing, would still be generally less than a day.

On the other end of the orbital transfer spectrum is the transfer maneuver associated with vehicles having thrust-to-weight ratios below 0.001. These classical, extremely low thrust-to-weight orbit transfers are characterized by a continuous burn spiral trajectory. Although this type of trajectory represents the shortest trip time for low thrust-to-weight propulsion system it also demands the greatest energy expenditure. The mission velocity in this regime is 19,200 fps, and the value remains essentially independent of vehicle thrust-to-weight ratio. The low thrust-to-weight solar rocket system results in trip times in excess of 10 days.

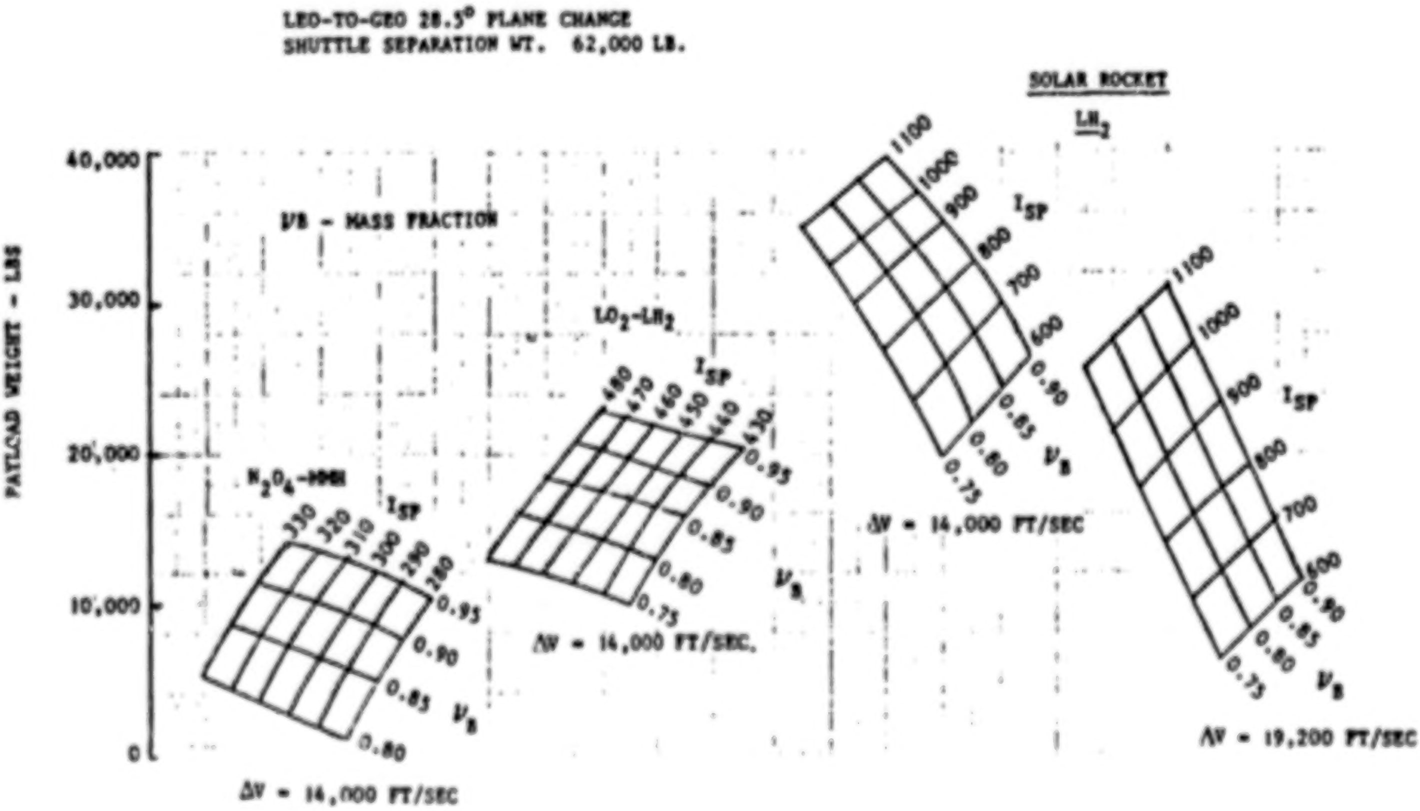
DELTA-V REQUIREMENTS VERSUS THRUST TO WEIGHT



Payload Capability for Various Propulsion Systems

The payload delivered to GEO by chemical systems and the solar rocket using LH_2 are shown for a range of stage mass fractions and typical ranges in their respective specific impulses. It is clearly seen that the solar rocket at the higher velocity requirements of 19,200 ft/sec must have specific impulses in excess of 800 Secs in order to improve performance over the cryogen propulsion stages (LO_2-LH_2).

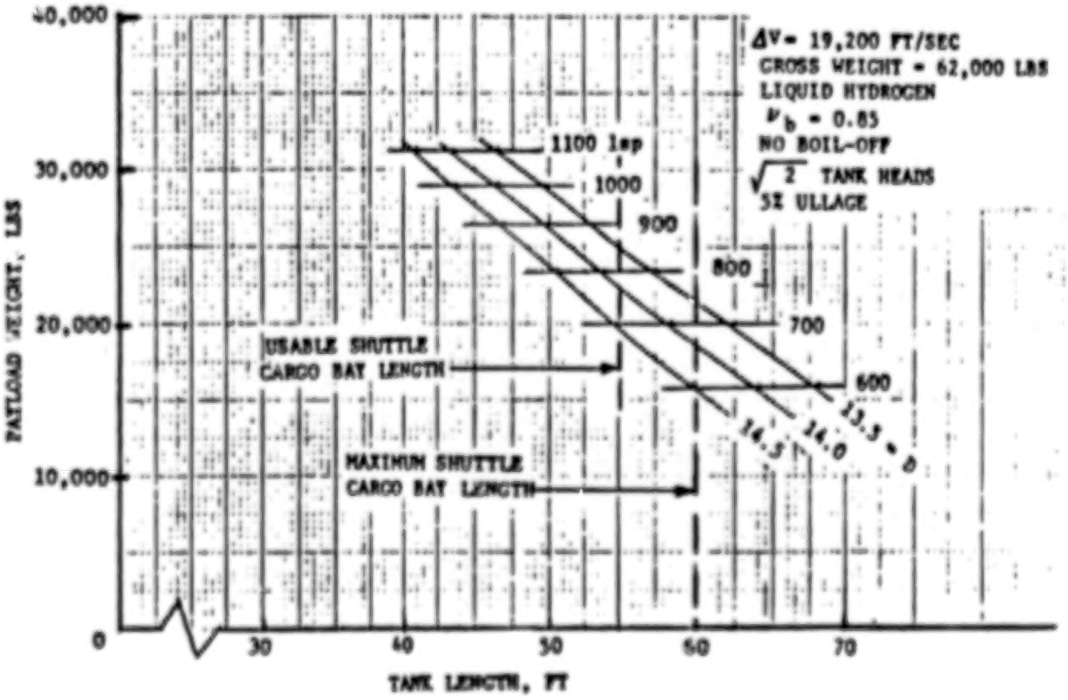
PAYLOAD CAPABILITY FOR VARIOUS PROPULSION SYSTEMS



Payload as a Function of Tank Geometry and Specific Impulse

The nominal dimensions of Shuttle cargo bay are 15 feet in diameter and 60 feet long. The LH_2 propellant will require the use of multilayer insulation systems, and an allowance for cradle thickness must also be made. Tankage inside diameters of 13.5, 14.0, and 14.5 feet have been assumed. The usable length of the cargo bay is 56 feet to allow for clearance and removal for the bay. Because of the low density of liquid hydrogen (4.4 lb/ft^3) the tank volume required to hold the quantity of propellant consistent with a 62,000-pound separation weight may exceed the usable volume of the cargo bay. The length of the hydrogen tank required as a function of diameter for the 62,000-pound separation weight constraint, shows that the vehicle tends to be limited by the orbiter's volume constraints.

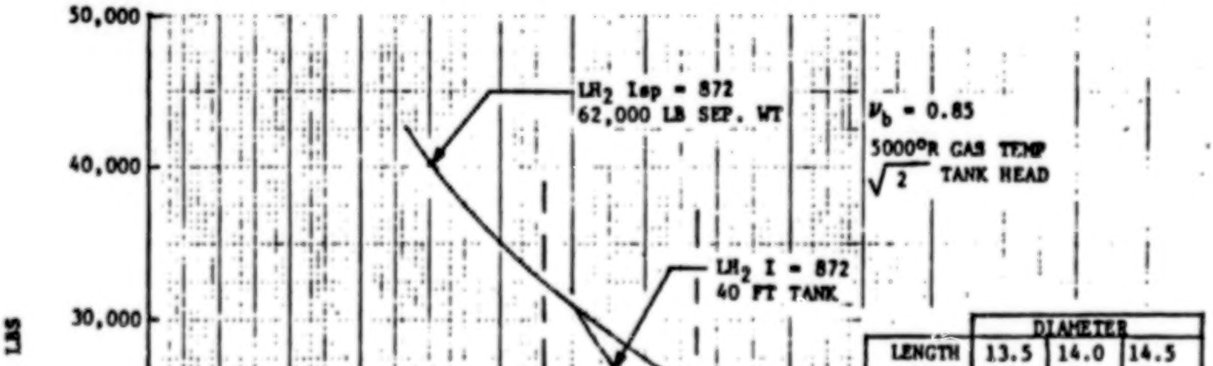
PAYLOAD AS A FUNCTION OF TANK GEOMETRY AND SPECIFIC IMPULSE



Payload Weight as a Function of ΔV - LH_2 and NH_3

Compared on this chart are the relative performance of two fuels used for the solar rocket. The denser NH_3 is not limited by the orbiter's cargo bay volume for the higher velocity increments, but with it's lower I_{sp} =440 secs has lower payload delivery capability than the LH_2 system constrained to a 40 foot long tank. This length will allow bay length to include the thruster, collectors and payload envelopes.

PAYLOAD WEIGHT AS A FUNCTION OF ΔV - LH_2 AND NH_3



Types of Transfer Maneuvers

The basic mission identified earlier consists of transferring a payload from LEO to GEO. Depending on the thrust-to-weight ratio of the orbit transfer vehicle, the transfer maneuvers can be generally divided into three distinct types. The mission velocity requirements range from a low of 14,000 fps to a high of 19,200 fps, depending on the vehicle thrust-to-weight ratio of the orbit transfer vehicle. It is recognized that continuous thrusting is not possible in low earth orbit due to eclipse periods. The descriptor "continuous" should be interpreted to mean that thrusting occurs whenever solar energy is available. In previous studies, it was found that the inclusion of the time spent traversing the Earth shadow results in a trip-time increase of approximately 10 percent at no increase in propellant expended.

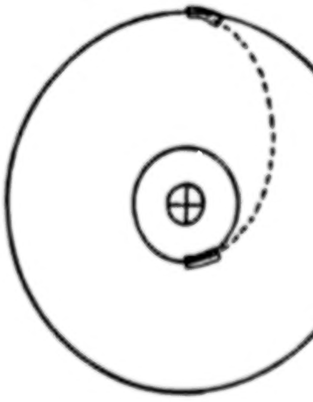
A viable alternative to the classical continuous burn spiral transfer method is to perform the burns only in the vicinity of perigee and/or apogeee. Theoretically, with an infinite number of impulses,it should be possible to reduce the required mission velocity to that attained from purely impulsive burns.

TYPES OF TRANSFER MANEUVERS

TWO IMPULSE

ONE PERIGEE BURN
ONE APOGEE BURN

T/W > 0.01



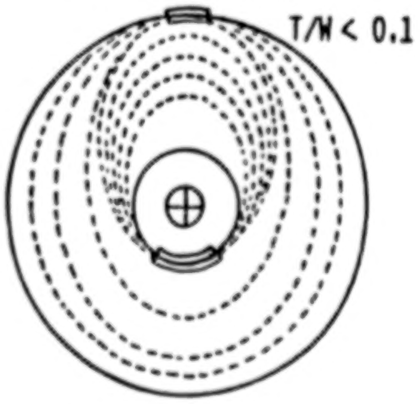
LEO TO GEO

14000 ≤ ΔV   17000 FPS

TRIP TIME < DAY

MULTI IMPULSE

MORE THAN ONE PERIGEE
BURNS AND MORE THAN
ONE "INSERTION" BURNS
NEAR FINAL APOGEE



LEO TO GEO

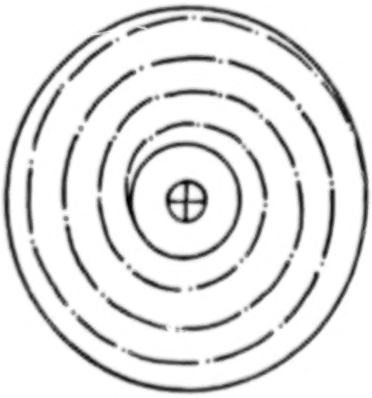
14000 ≤ ΔV ≤ 19200 FPS

TRIP TIME > DAYS

CONTINUOUS BURN

SPIRAL TRAJECTORY

T/W < 0.001



LEO TO GEO

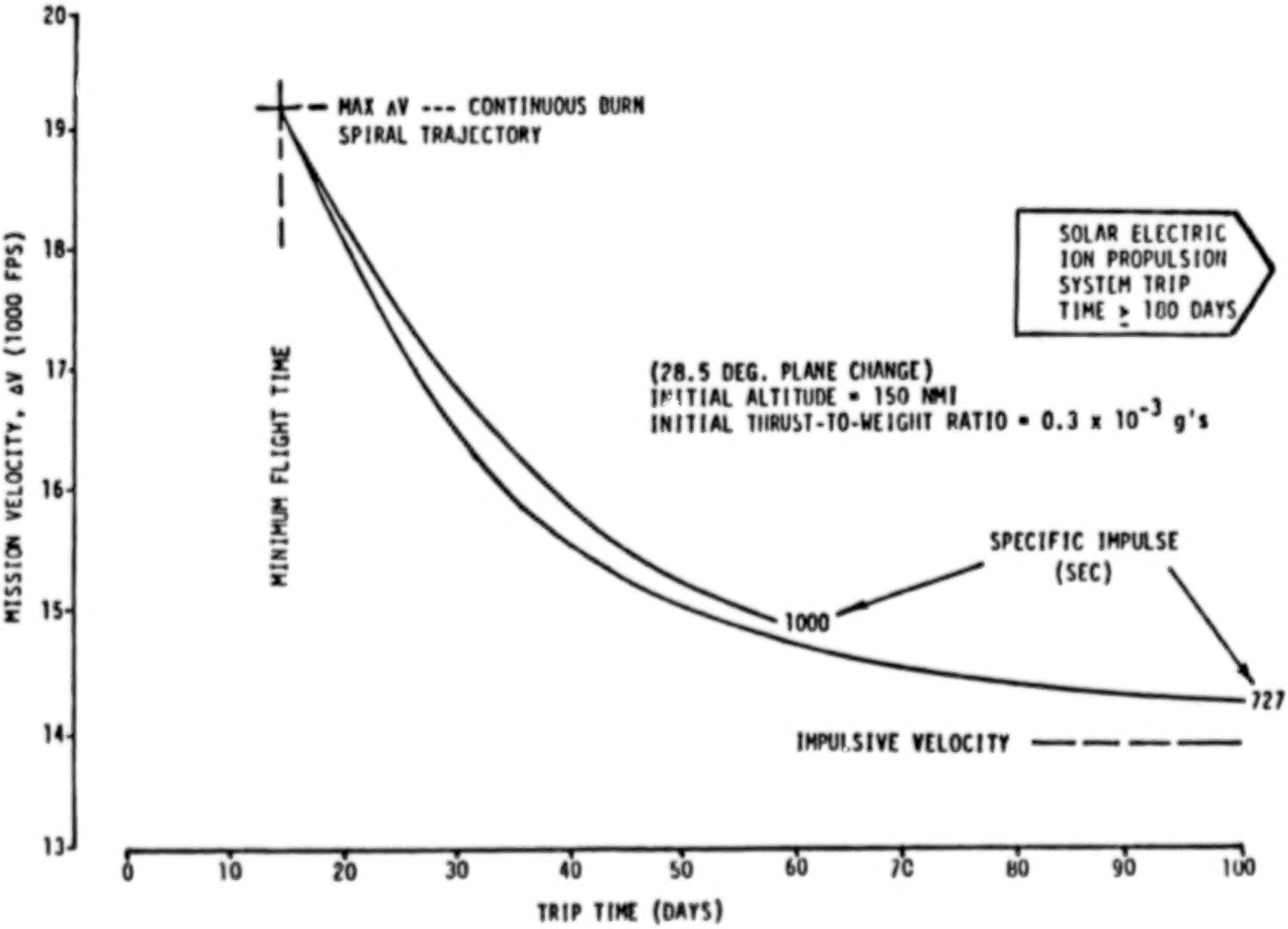
ΔV ≈ 19200 FPS

TRIP TIME > DAYS

Delta V as a Function of Trip Time - LEO to GEO

The relationship between the mission ΔV and corresponding time obtained by optimizing the multiburn transfer is illustrated. The example is for an initial thrust-to-weight of 0.3×10^{-3} g's and two representative specific impulse values (727 and 1000 sec). Thus, for example, by extending the transfer from 14 days to 30 days, the mission ΔV can be reduced from 19,200 fps to 16,500 fps (Isp =727 sec). These trip time increases should, however, be considered in relationship to the 180+ trip times that are characteristic of the mercury ion bombardment propulsion systems.

DELTA V AS A FUNCTION OF TRIP TIME - LEO TO GEO

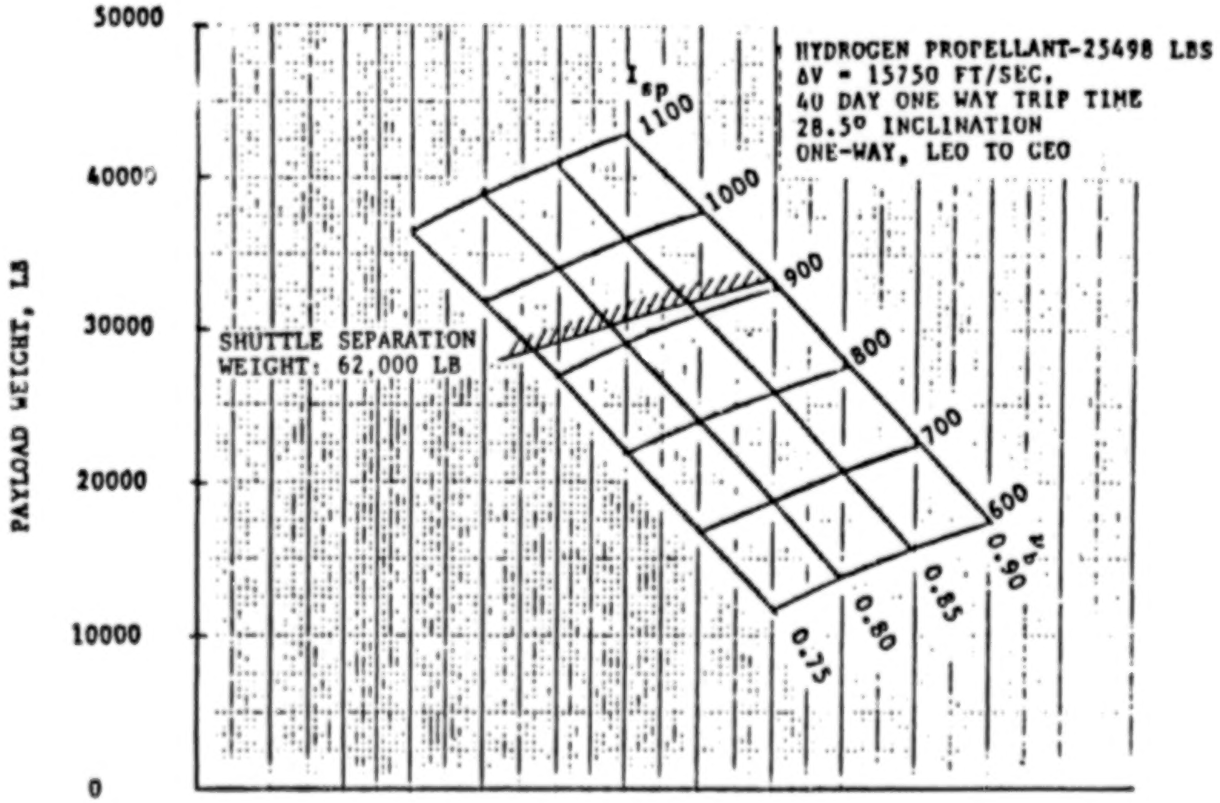


Payload Weight as a Function of Specific Impulse and Mass Fraction (14.5 ft diameter, 40 ft Tank)

To illustrate the effect of trip time, a carpet plot for a one-way 40-day trip, was prepared; this is presented. It may be seen that the 40-day trip time payload is 29,000 pounds ($I = 872$ $\mu_b = 0.85$) and is 8500 pounds greater than for the 14-day case with continuous burn. The decision as to whether an 8500 pound payload increase is desirable in terms of a 26-day increase in trip time must be made by the mission planner.

It is seen that the payload capacity for the higher specific impulses will be limited by the Shuttle separation weight of 62000 pounds for the 40 day mission with the multiple impulsive trajectory.

PAYLOAD WEIGHT AS A FUNCTION OF SPECIFIC IMPULSE AND MASS FRACTION
(14.5 FT DIAMETER, 40-FT TANK)



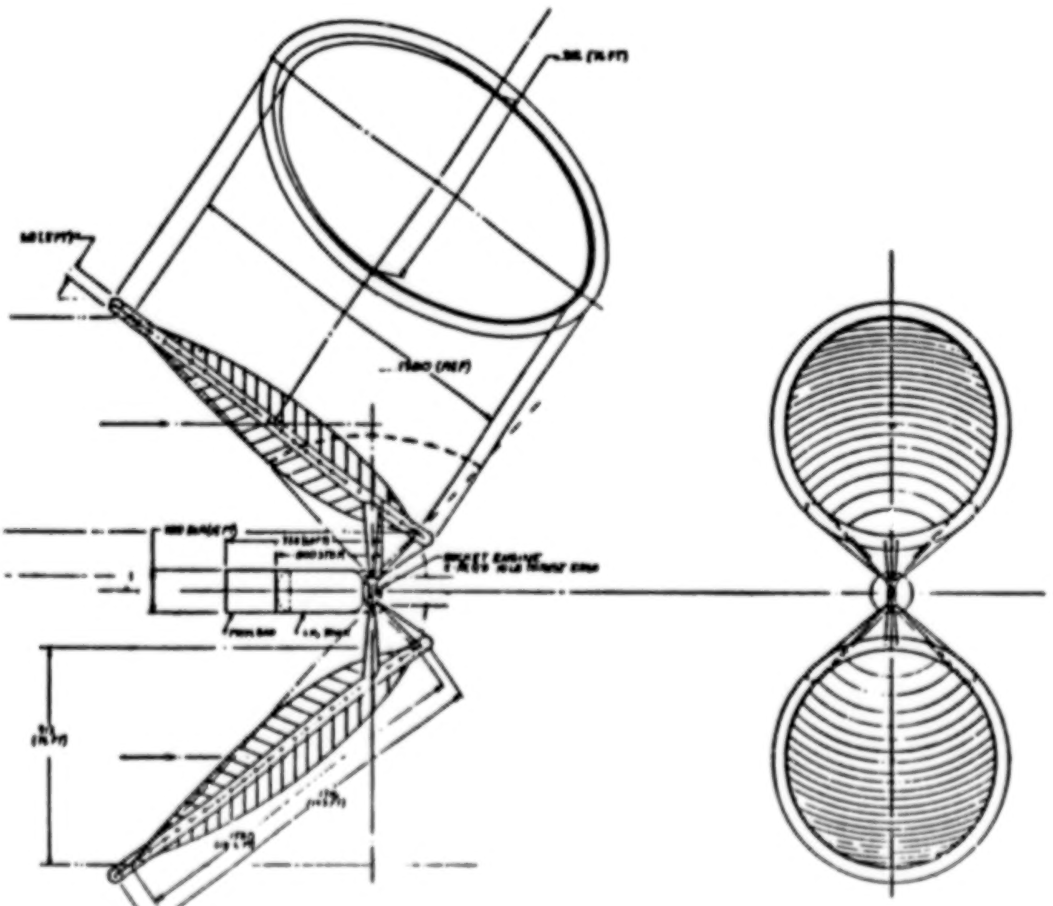
Off-Axis Parabaloid Concentrator Configuration

The basic operating principal of the solar rocket is the use of solar energy to heat a working fluid. The solar collector concentrates the energy through the absorber's window wherein the working fluid is heated to temperatures in excess of 5000°R and the hot gases are expelled via the thruster nozzles.

The primary requirements of a solar collector for a solar rocket system are deployability, low specific mass, and high concentration ratio. The latter is necessary to achieve high temperature and specific impulse of the heated propellant. Of the various candidates considered, only an inflated, non-rigidized concentrator design meets these requirements. The pressure required to maintain the surface contour accuracy is extremely low that any likely puncture of the collector membrane by micrometeoroids encountered during the transfer mission, will allow relatively small volume of gas to escape. (about 200 pounds/mission).

The solar tracking and tangential thrusting can be accomplished by providing a single degree of rotation of the parabolic collectors about an axis normal to vehicle's center line and the second degree is obtained by rotation of complete vehicle about its roll axis.

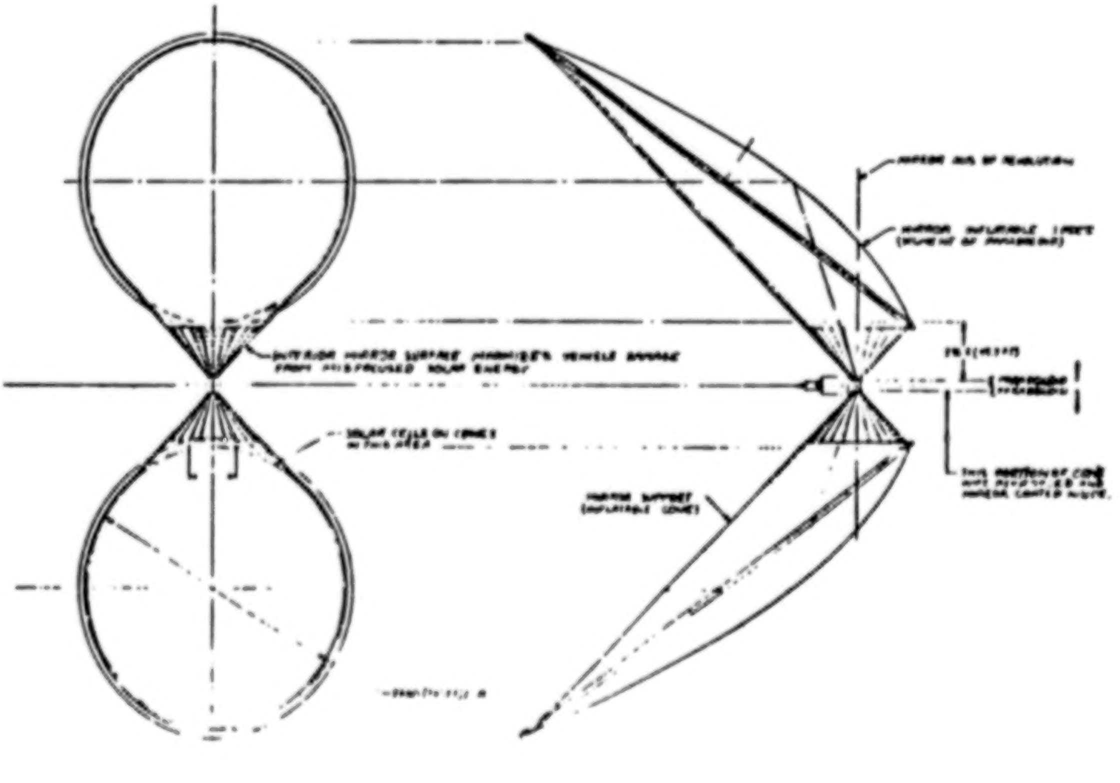
OFF-AXIS PARABALOID CONCENTRATOR CONFIGURATION



Inflatable Cone/Parabaloid Collector

Design concept is a high thrust vehicle with a parabaloid collector of higher concentration ratio. The inflatable mirror surface is a segment of a parabaloid, while the interior surface is an inflatable cone segment.

INFLATABLE CONE/PARABOLOID COLLECTOR

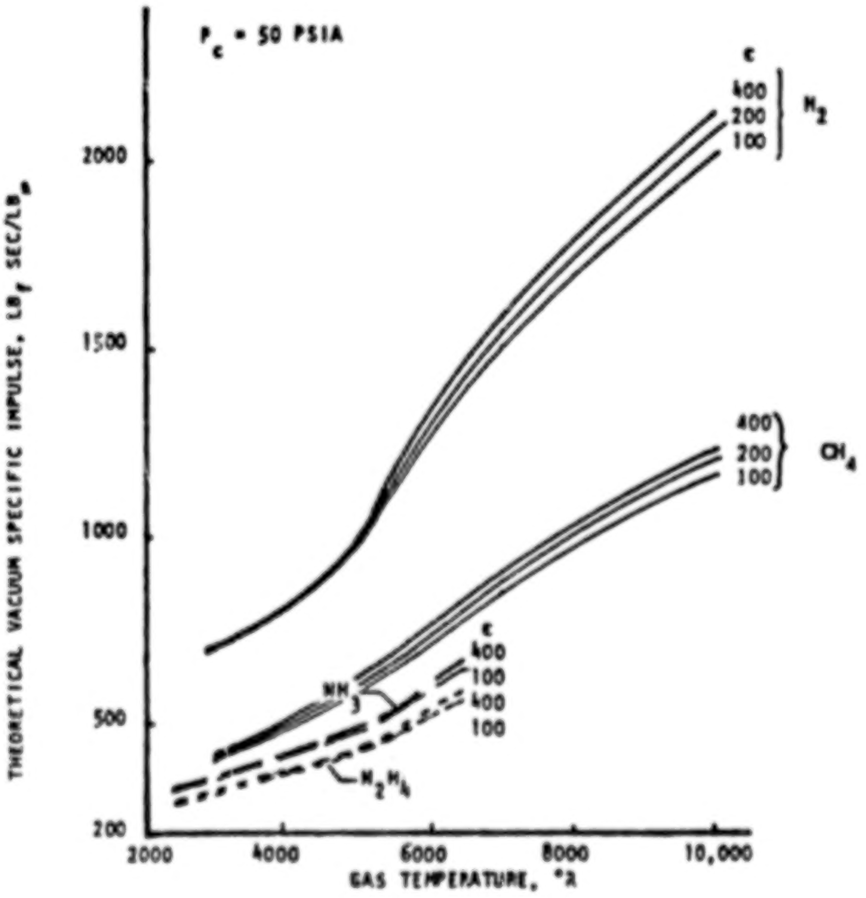


Theoretical Vacuum Specific Impulse Variation with Gas Temperature for Hydrogen, Hydrazine, Ammonia and Methane

The variation of theoretical equilibrium (shifting) vacuum specific impulse with gas temperature was determined for H_2, CH_4, NH_3, and N_2H_4 at a chamber pressure of 50 psia, as shown. Data for thruster nozzle area ratios ranging from 100 to 400 are presented. For a given propellant gas temperature, H_2 achieved a theoretical specific impulse a factor of two higher than that of NH_3 or N_2H_4 and approximately 77-percent higher than of CH_4. The increase in slope of specific impulse versus temperature with hydrogen at approximately 5000°R is the result of an increase in the amount of dissociated hydrogen. Methane specific impulse values for a given temperature were 14 to 24 percent higher than that of NH_3. As shown, the variation of theoretical specific impulse for an area ratio increase from 100 to 400 was approximately six percent at 7000°R gas temperature for H_2.

The desired high propellant temperatures represent a problem for CH_4. Above 1760°R, CH_4 starts to decompose and forms coke, which deposits on coolant passage walls. This coking layer acts as an insulating layer and makes cooling of the heated surface difficult. Therefore methane was not considered a potentially attractive propellant for the solar rocket.

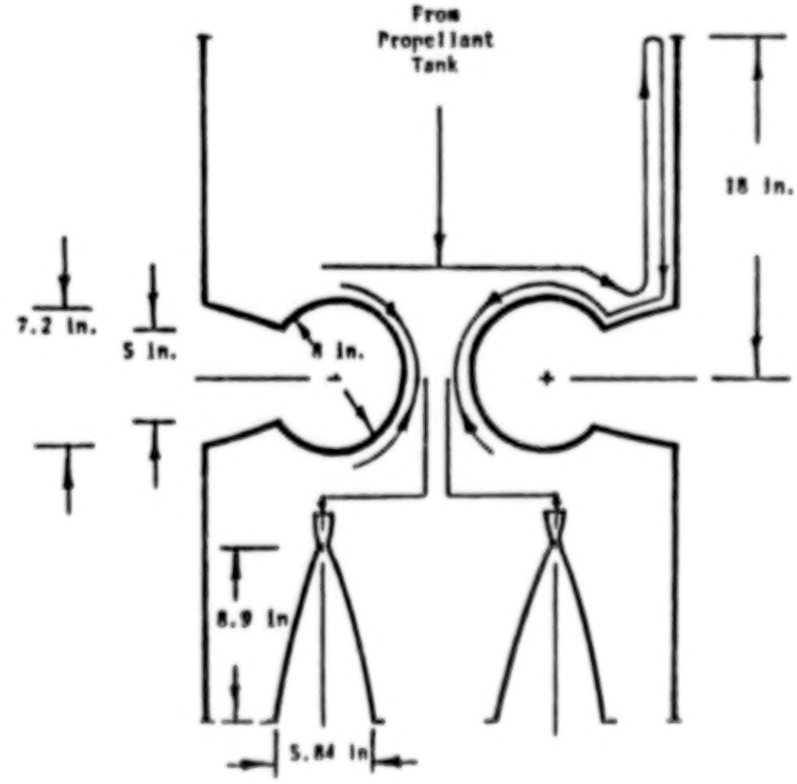
THEORETICAL VACUUM SPECIFIC IMPULSE VARIATION WITH GAS TEMPERATURE FOR HYDROGEN, HYDRAZINE, AMMONIA AND METHANE



Heat Exchanger Cavity Absorber/Thruster (Two Thrusts) (Hydrogen at 5000°R)

A heat exchanger cavity absorber/thruster configuration with hydrogen at 5000°R (highest performance) consists of a reflector cone (Winston horn) with a 7.2-inch-diameter inlet, an 8-inch-diameter sphere to absorb the reflector cone magnified heat flex, and a 36-inch-diameter annular disc absorber. This sphere/horn/disc absorber configuration can achieve a 71-percent overall efficiency. The two thruster, two absorber configuration at a chamber pressure of 50 psia will deliver a specific impulse of 861 lb f/sec and a thrust of 43 lbf/ The nozzle exit is placed at the same plane as the edge of the flat disc to prevent plume impingement on the disc absorber.

HEAT EXCHANGER CAVITY ABSORBER/THRUSTER (TWO THRUSTERS) (HYDROGEN AT 5,000°R)

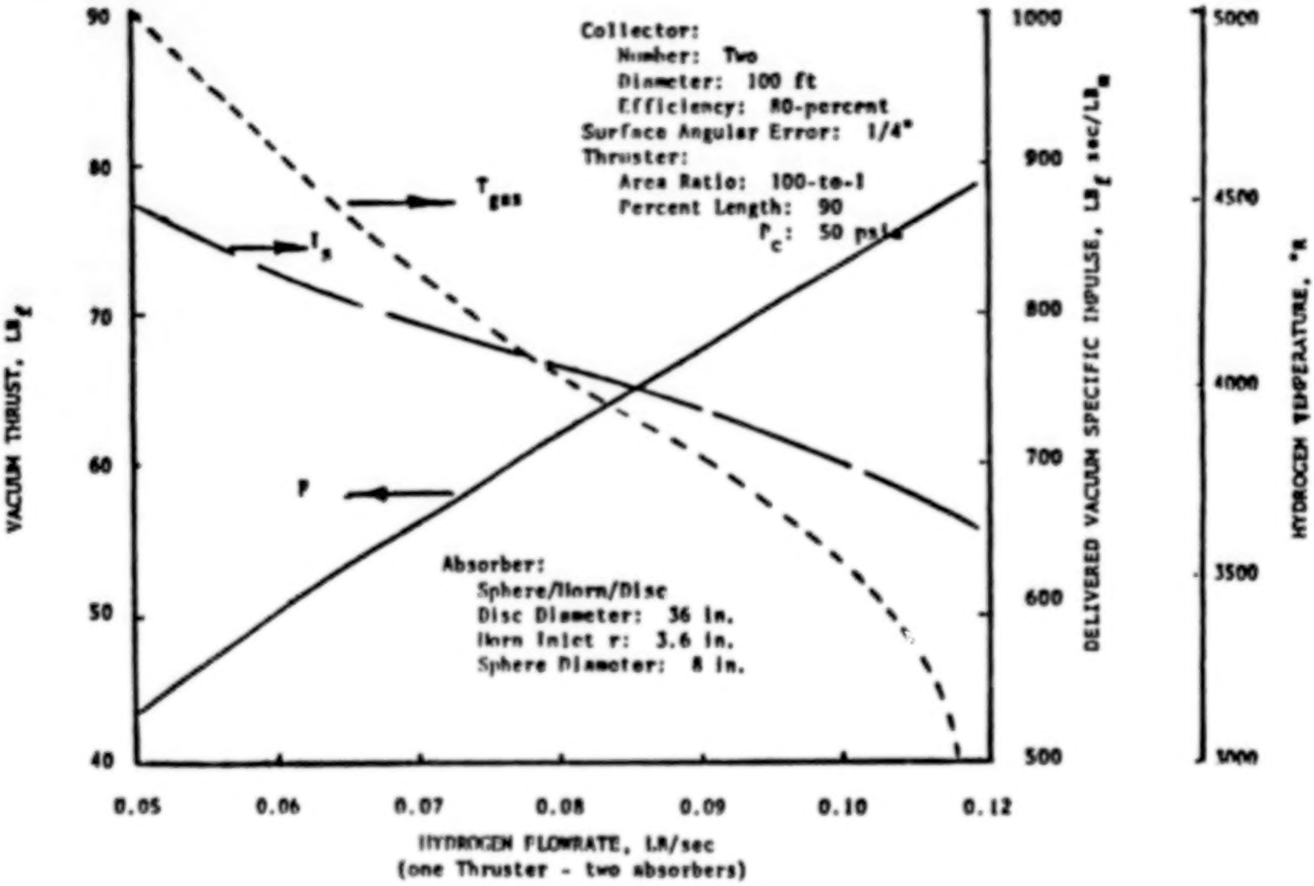


Collector:
- Number: Two
- Diameter: 100 ft
- Efficiency: 80-percent
- Surface Angular Error: 1/4-degree

Absorber:
- Spherical/ Horn/Disc
- Disc Diameter: 36-in.
- Horn Inlet r: 3.6-in.
- Sphere Diameter: 8-in.
- ϵ_{sphere}: 0.3
- ϵ_{disc}: 0.9
- Efficiency: 71-percent

Thruster:
- Throat Diameter: 0.584-in.
- Area Ratio: 100-to-1
- Chamber Pressure: 50 psia
- Flowrate: 0.025 lb/sec each
- Thrust: 21.5 lb_f each
- Special Impulse: 861 lb_f sec/lb_m

HYDROGEN HEAT EXCHANGER ABSORBER/THRUSTER PERFORMANCE

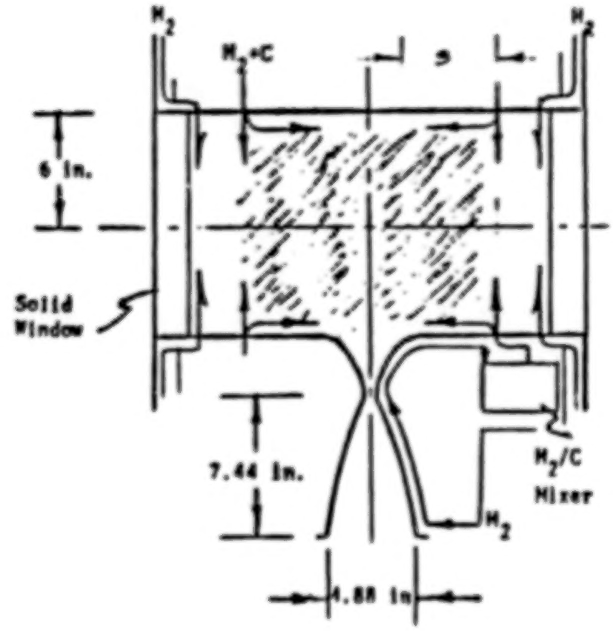


Particulate Absorber/Thruster (Hydrogen/Carbon at 7000°R)

A similar system as analyzed with a 100-to-1 area ratio, 90-percent length bell nozzles with two 100-foot-diameter collectors and using Hydrogen/carbon (10-percent) as the propellant. For the 6000°R-to-8000°R propellant temperature range evaluated, the delivered specific impulse varied from 940 lb_f sec/lb_m to 1100 lb_f sec/lb_m for the H_2/C propellant with a carbon mass fraction of 0.1. The thrust decreased from 23.5 lb_f to 9 lb_f as the propellant temperature was increased from 6000 to 8000°R.

A particulate absorber/thruster configuration with H_2/C at 7000°R consists of a 6-inch-radius cylinder plus an annular disc. Hydrogen first cools the annular disc absorber, then splits (1) to cool the solid window and (2) to cool the thruster and absorber body. Once the absorber body is cooled, the H_2 enters a solid-particle gas mixer, and the H_2C mixture is injected downstream of the window. The cylindricial particulate absorber/disc configuration achieved a 51-percent overall efficiency using the optimistic absorber analysis approach the single thruster at a chamber pressure of 50 psia resulted in a delivered specific impulse of 1041 lb_f sec/lb_m and thrust of 14 lb_f.

PARTICULATE ABSORBER/THRUSTER
(HYDROGEN/CARBON AT 7000°R)

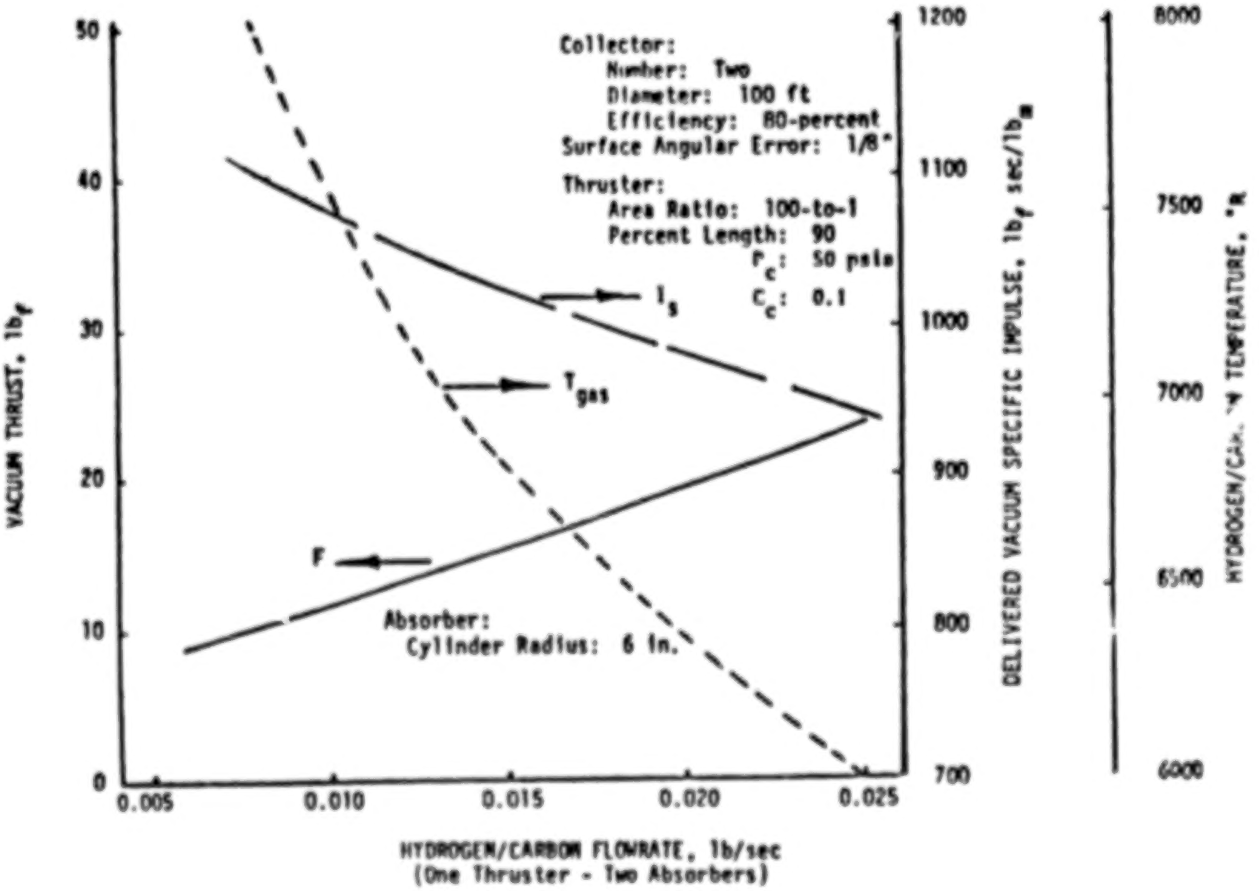


Collector:
  Number: Two
  Diameter: 100 ft
  Efficiency: 80-percent
Surface Angular Error: 1/8-degree

Absorber:
  Cylindrical Particulate/Thruster/Disc
    ϵ_{gas}: 1.0
    ϵ_w: 0.1
    η_w: 0.9
  Efficiency: 51-percent

Thruster:
  Throat Diameter: 0.489 in.
  Area Ratio: 100-to-1
  Chamber Pressure: 50 psia
  C_c: 0.1
  Flowrate: 0.013 lb/sec
  Thrust: 14 lb_f
  Specific Impulse: 1041 lb_f sec/lb_m

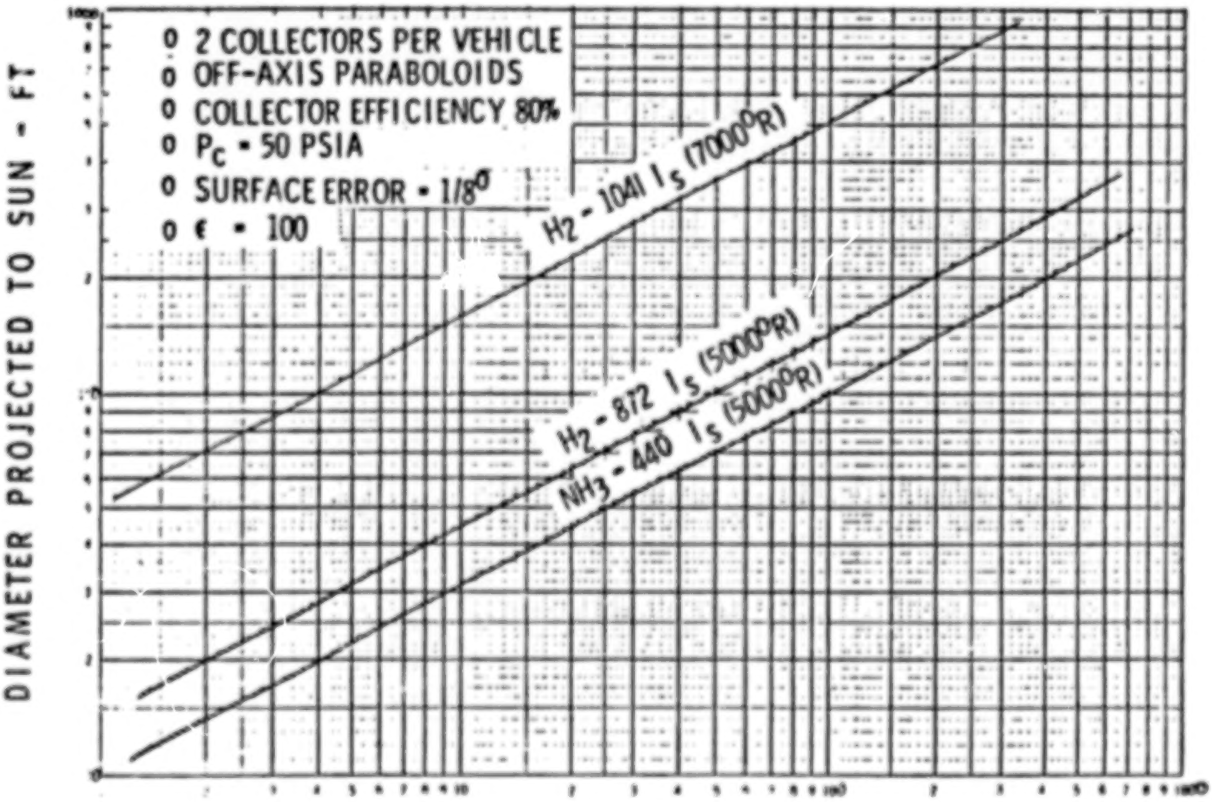
HYDROGEN/CARBON PARTICULATE ABSORBER/THRUSTER PERFORMANCE



Required Concentrator Diameter

The diameter of the solar collector is dependent on the thrust level required and the concentration ratio necessary to attain the desired cavity temperatures. Based on collector efficiency of 80% and a RMS surface error of 1/8 the required diameter for each collector is shown in this chart.

REQUIRED CONCENTRATOR DIAMETER



TOTAL THRUST - LB

Sundstrand/Goodyear Collector Experience

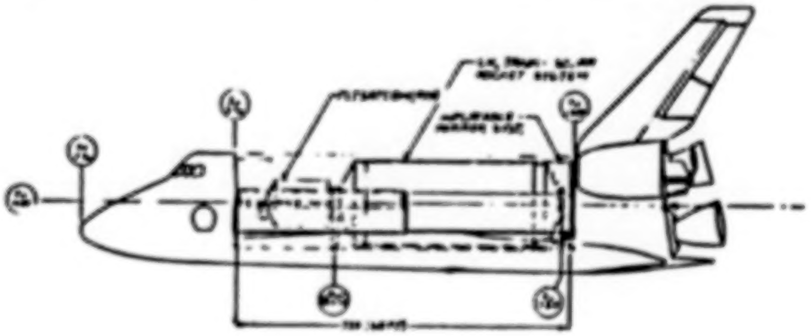
During the mid 1960 several light weight collectors were fabricated to determine collector surface accuracy and performance.

The 44.5 foot-diameter concentrator built under the ASTEC program used a foam rigidized aluminized mylar concept which demonstrated a concentration ratio of 3200. The contour accuracy was within ± 0.25 inch (equivalent to ± 0.10° surface error standard deviation). Subsequent analysis of the concentrator indicated that the foam caused distortions in the concentrator surface which caused a reduction in the potentially available concentration ratio. The estimated concentration ratio used in the study was 9800. Through the use of Winston horn (compound parabolic reflector skirt), an average concentration ratio at the exit of the horn of 14328 is expected.

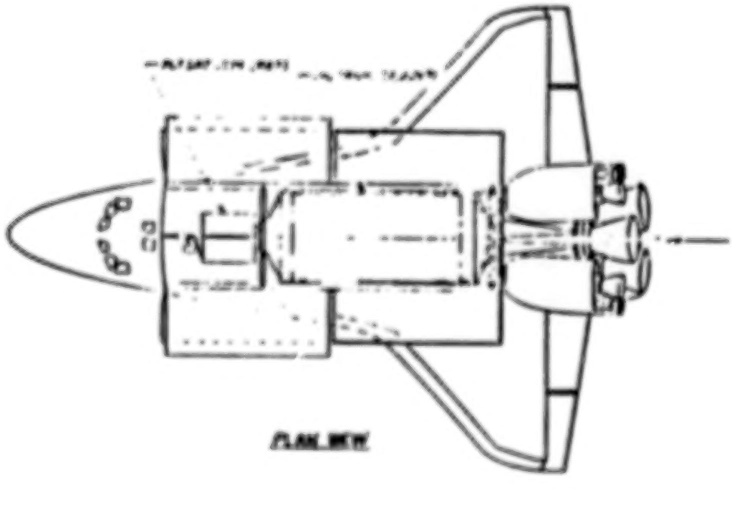
SUNDSTRAND/GOODYEAR COLLECTOR EXPERIENCE

- SUNDSTRAND WAS CONTRACTOR IN MID-1960's FOR PROJECT ASTEC (15 KW SOLAR POWER SYSTEM)
- CONCENTRATOR WAS SUBCONTRACTED TO GOODYEAR
- INFLATED AL-MYLAR, FOAM RIGIDIZED DESIGN
- 10 FT. DIA. MODEL - 3900 C.R.
- 44.5 FT. DIA. MODEL - 3200 C.R.
- CONTOUR ACCURACY OF 44.5 FT. MODEL WAS WITHIN ± 0.25" (EQUIVALENT TO <± 0.1° SURFACE ERROR STD. DEVIATION)
- SUNDSTRAND SAYS NON-RIGIDIZED DESIGN IS MUCH BETTER THAN RIGIDIZED FOR HIGH ACCURACY MIRRORS.
- INDICATIONS ARE THAT 1/8° SURFACE ERROR CAN BE ACHIEVED IN SPACE (SEARCHLIGHT QUALITY)

SOLAR ROCKET SYSTEM
SHUTTLE LAUNCH INSTALLATION



SIDE VIEW



PLAN VIEW

Parametric Synthesis

The parametric analysis of the solar rocket system was achieved using the Solar Thermal Orbital Propulsion—Computerized Unmanned Spacecraft Synthesis program (STOP CUSS). This program allows the investigation of various design and subsystem parameters and how these parameters affect the overall vehicle performance.

The major structural elements of the propulsion stage are the propellant tankage, the solar collector components, and the thruster system. Weight allowances must be assigned to each of these major elements to account statistically for the secondary structure and ancillary equipment. Each of the structural components is divided into its element models, each element is defined analytically, and a preliminary design synthesis is conducted on the individual elements to identify minimum weights and scaling laws for feasible designs. A correlation factor (non-optimum weight, etc.) is applied to these laws based on historical data pertinent to the type of material, construction, and complexity of the component.

The synthesis approach starts with the sizing of the tanks to contain the propellant used for propulsive changes in the vehicle's orbit (LEO to GEO, etc) and the propellant that will boil-off during the longer trip times. The heating rate and total heat input throughout the various mission trajectory segments will influence the propellant boiled-off.

The quantity of propellant boil-off is a function of the vehicle's thrust-to-weight (hence trip time), the surface area of the tank(s) exposed to the thermal environment, and the tank insulation concepts. Sizing and number of propellant tanks employed for the large payload designs are dictated by the Shuttle orbiter's cargo bay physical limitations.

PARAMETRIC SYNTHESIS

SOLAR THERMAL ORBITAL PROPULSION
COMPUTERIZED UNMANNED SPACECRAFT
SYNTHESIS
(STOP CUSS)

EFFECTS OF:

- PAYLOAD SIZE
- INSULATION THICKNESS
- THRUST-TO-WEIGHT
- SPECIFIC IMPULSE
- SHUTTLE CONSTRAINTS
- MISSION TRIP TIME
- TANK PRESSURES

Effect of Insulation Thickness

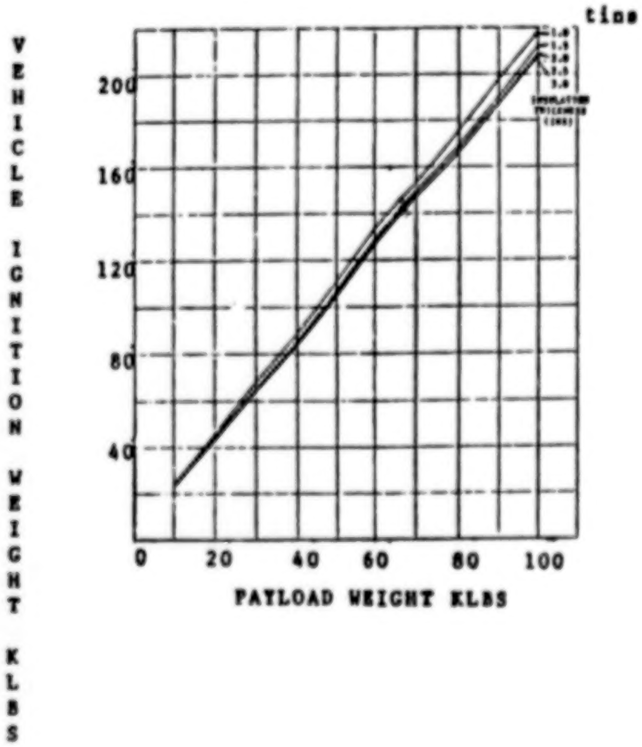
Results show that for the LEO-GEO and LEO-GEO and return trips the Multilayer insulation should be about 1.5 inches thick to preclude too much hydrogen boil-off during the multi-day trip time.

EFFECT OF INSULATION THICKNESS

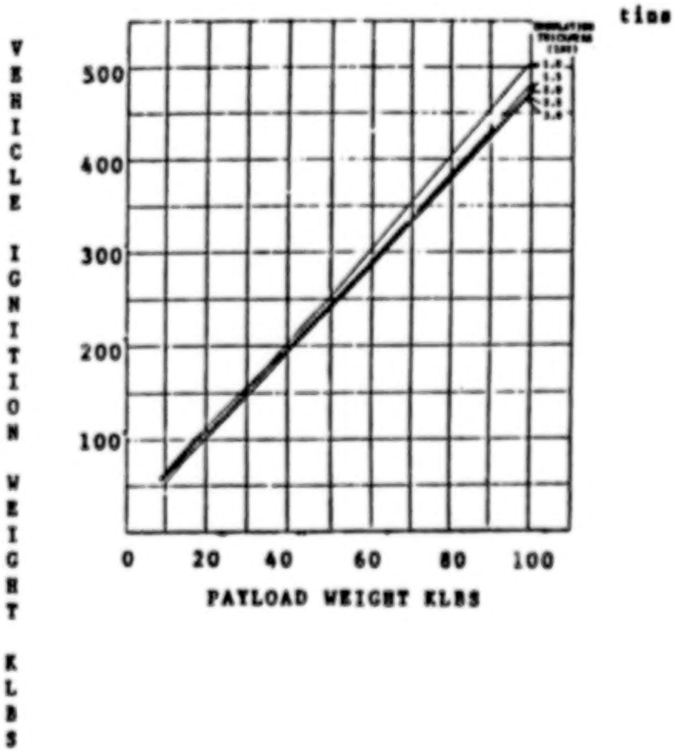
SPECIFIC IMPULSE = 1041 SECS

T/W = 0.00003

LEO - GEO



LEO - GEO & RETURN

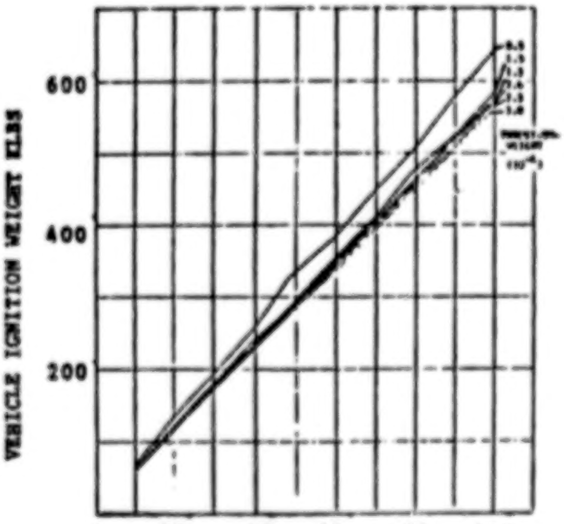


Effect of Thrust-to-Weight for LEO-GEO and Return

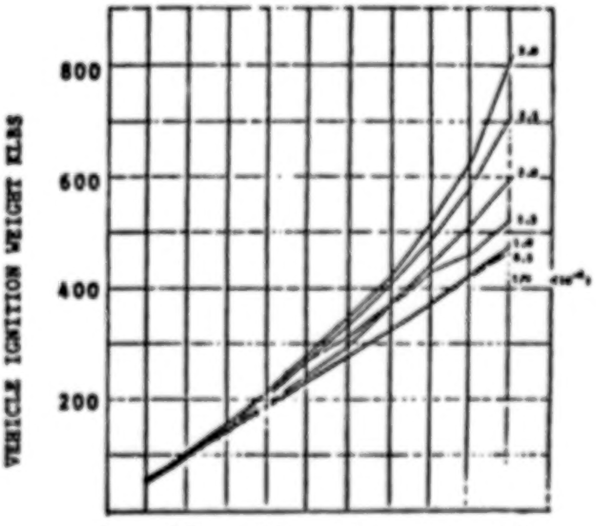
It is interesting to note that for the low specific impulse (872 sec) system that payload performance is improved by increasing the T/W from 0.5 to 3.0 x 10^{-4}. For the higher specific impulse (1041 secs) the opposite is true. There can be significant decreases in payload performance for the higher T/W at the larger payload ranges. This is due to the larger size solar collectors required to obtain the 7000°R temperatures, wherein the collector weight becomes a significant percent of the stage empty weight.

EFFECT OF THRUST TO WEIGHT FOR LEO-GEO AND RETURN

SPECIFIC IMPULSE 872 SEC



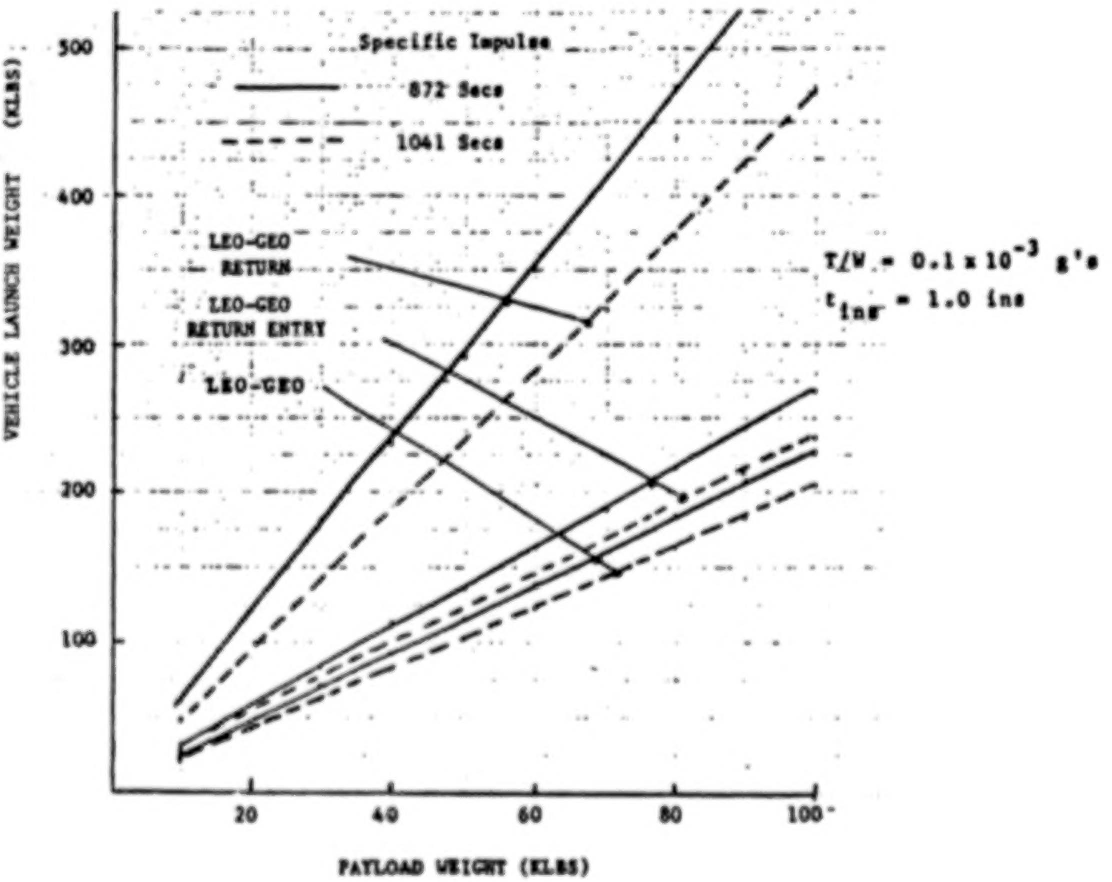
SPECIFIC IMPULSE 1041 SEC



Effect of Improved Engine Performance

This chart shows the vehicle initial launch weight required for payloads ranging from 10,000 lbs to 100,000 lbs. Three missions are considered, these being, expendable LEO to GEO, recoverable LEO-to-GEO thirty days stay at GEO and then return only the vehicle stages and thirdly the mission which recovers both the stage and a payload with a thirty day stay at GEO.

EFFECT OF IMPROVED ENGINE PERFORMANCE



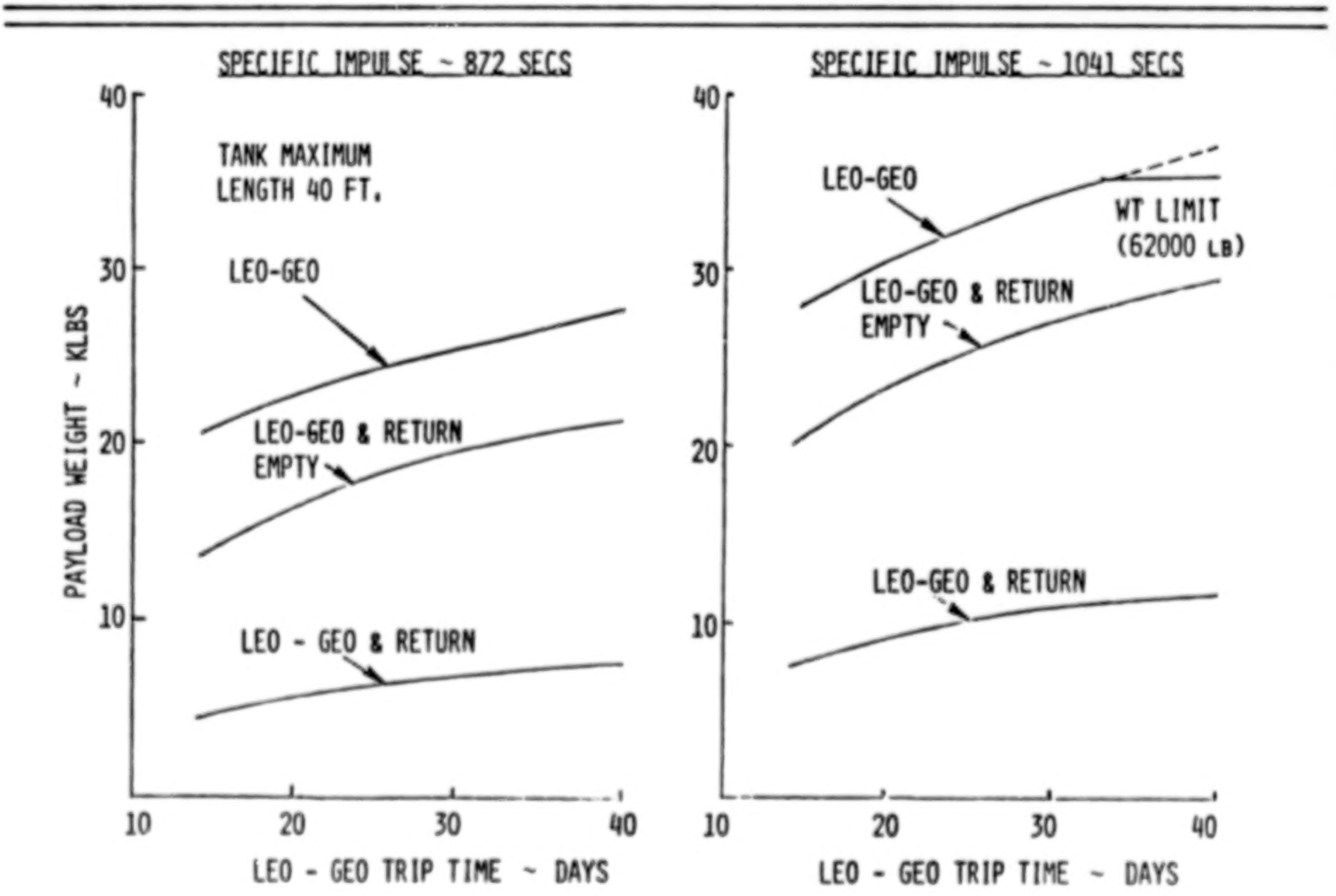
Single Orbiter Launch Capability

The single Shuttle launch payload capability increases as the trip duration increases. The trajectories considered for these increased flight times are for the apogee/perigee burn flight modes which significantly reduced the total velocity requirements. The velocity required is 19,200 ft/sec at the 14-day trip time reducing to 15,750 ft/sec for the 40-day duration. The extended mission duration has the effect of increasing the amount of propellant boiled-off, which negates some of the benefits of the reduction in velocity requirements.

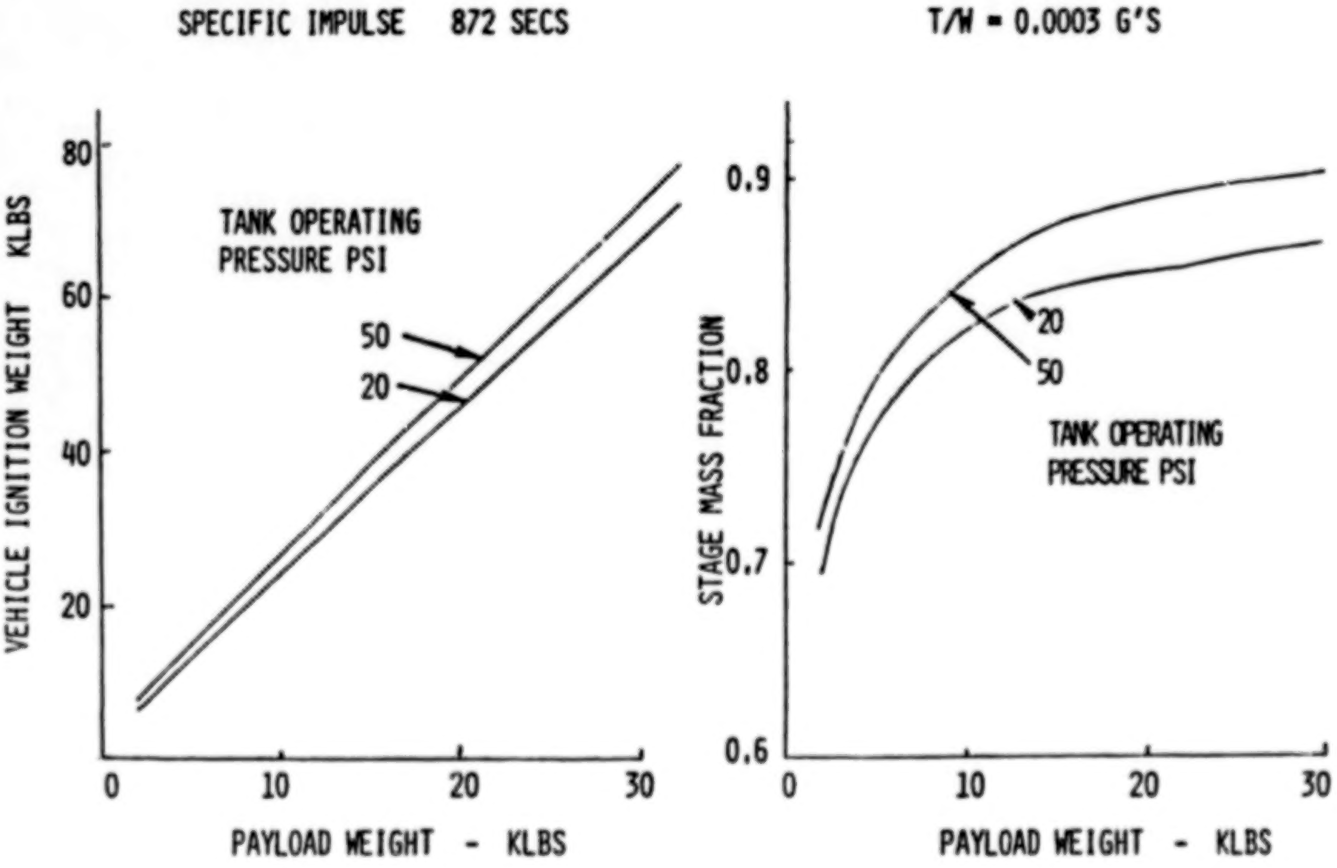
For the LEO-to-GEO mission, the payload delivered by an orbiter launch vehicle ranges from 22,000 to 27,000 pounds for the low-temperature (5000°R) thruster system. This payload can be increased by 20 percent if the high-temperature (7000°R) thruster is used for the propulsion system.

Missions which return the vehicle but leave the payload at GEO can place payloads of from 15,000 to 20,000 pounds into the geosynchronous orbit. This type of mission does not benefit from the improved thruster performance of the high-temperature system. The payload is very sensitive to the returned stage inert weight. The collector weight for the high-temperature system constitutes a significant percentage of the stage inert weight and negates the gains from the higher impulse.

SINGLE ORBITER LAUNCH CAPABILITY



EFFECT OF TANK PRESSURE



Technology Development Areas

Although there appears to be performance improvements with the high temperature system (7000°R) there are several major technical development areas to be investigated. The propellant is carbon doped hydrogen which will tend to deposit on the absorber's window and hence reduce the energy entering the absorber's cavity, thus cutting down its thermal efficiency. A film of hydrogen across the inside of the window could possibly reduce the deposition problem. The higher temperatures are pushing even further the material requirements, while the solar collector is larger than the 5000°R system with equal thrust levels.

The inflatable collectors with their high concentration rates although ground test articles have been fabricated, their packaging and automated deployment in space present areas of untested technology. The multiple-burn trajectory with its coast periods between burns will require a defocusing of the collector.

TECHNOLOGY DEVELOPMENT AREAS

THRUSTERS

- INCREASE PERFORMANCE - HIGHER TEMPERATURES
- AVOIDANCE OF CARBON DEPOSITION

COLLECTORS

- COLLECTOR OPTIMIZATION (FACETS, DESIGN, C.G.)
- NON-UNIFORM STRESS OF PARABALOIDAL MEMBRANE
- HIGH ACCURACY COLLECTOR FABRICATION TECHNIQUES
- SPECULAR REFLECTANCE OF METALIZED FILMS
- STRUCTURAL DYNAMICS & THERMAL DEFORMATIONS
- DEFOCUSING DURING COAST PERIODS

TANKAGE

- PUMP-FED VS. PRESSURE-FED PROPELLANTS
- HIGH PERFORMANCE INSULATION DESIGN

CONTROL

- OPTIMUM STEERING POLICY
- C.G. SHIFTING WITH TRACKING
- GIMBALED ENGINES VS. RCS JETS

Conclusions

The solar rocket system presents an interesting alternative whose performance is between the best chemical and the electric propulsion system. The thrust-to-weight is about 10^{-3} which would make them attractive as propulsion systems for large flexible space structures.

CONCLUSIONS

- THE 5000°R SOLAR ROCKET SYSTEM IS WITHIN THE CURRENT STATE-OF-THE-ART
- THE 5000°R SOLAR ROCKET SYSTEM PERFORMANCE IS SUPERIOR TO AN LO_2-LH_2 ORBIT TRANSFER VEHICLE FOR MULTI-DAY TRANSIT TIMES
- THE PAYLOAD OF THE 5000°R SOLAR ROCKET FOR THE PAYLOAD-UP SPACECRAFT DOWN CASE IS GREATER THAN THE CHEMICAL SYSTEM.
- THE 7000°R SOLAR ROCKET SYSTEM WILL REQUIRE A SIGNIFICANT DEVELOPMENT EFFORT BUT THE PAYOFF FOR THE SINGLE SHUTTLE LAUNCH CASE IS SIGNIFICANT.
- SOLAR ROCKET HAS POTENTIAL FOR HIGHER ENERGY ORBIT TRANSFER AT LOWER THRUST-TO-WEIGHT RATIOS USING EFFICIENT MULTI-DAY TRANSIT MANEUVERS

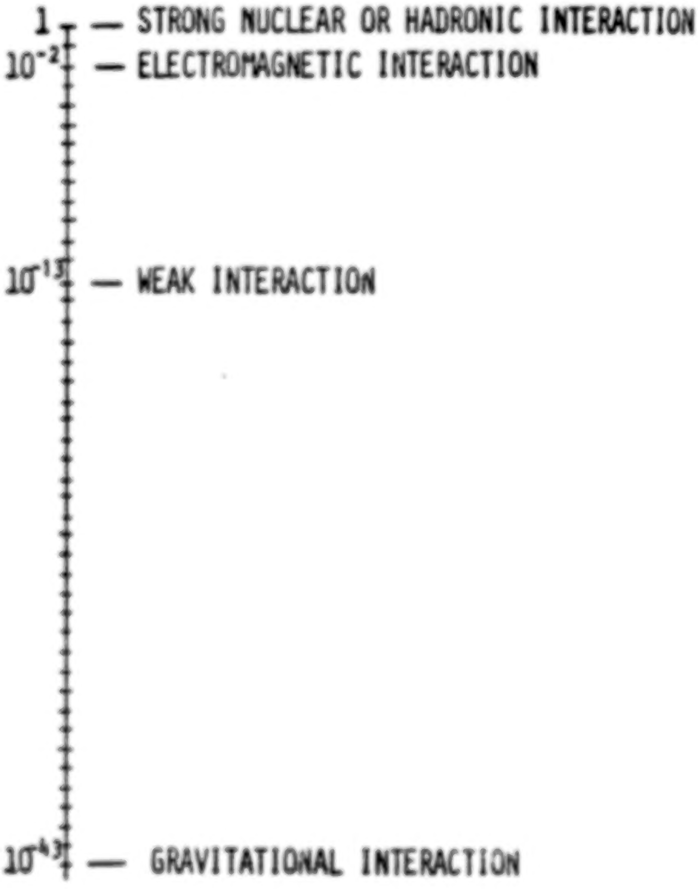
ADVANCED CONCEPTS

Bruce A. Banks
NASA Lewis Research Center

INTERACTIONS TO ENABLE PROPULSIVE FORCES

- STRONG NUCLEAR OR HADRONIC INTERACTIONS
- ELECTROMAGNETIC INTERACTIONS
- WEAK INTERACTIONS
- GRAVITATIONAL INTERACTIONS

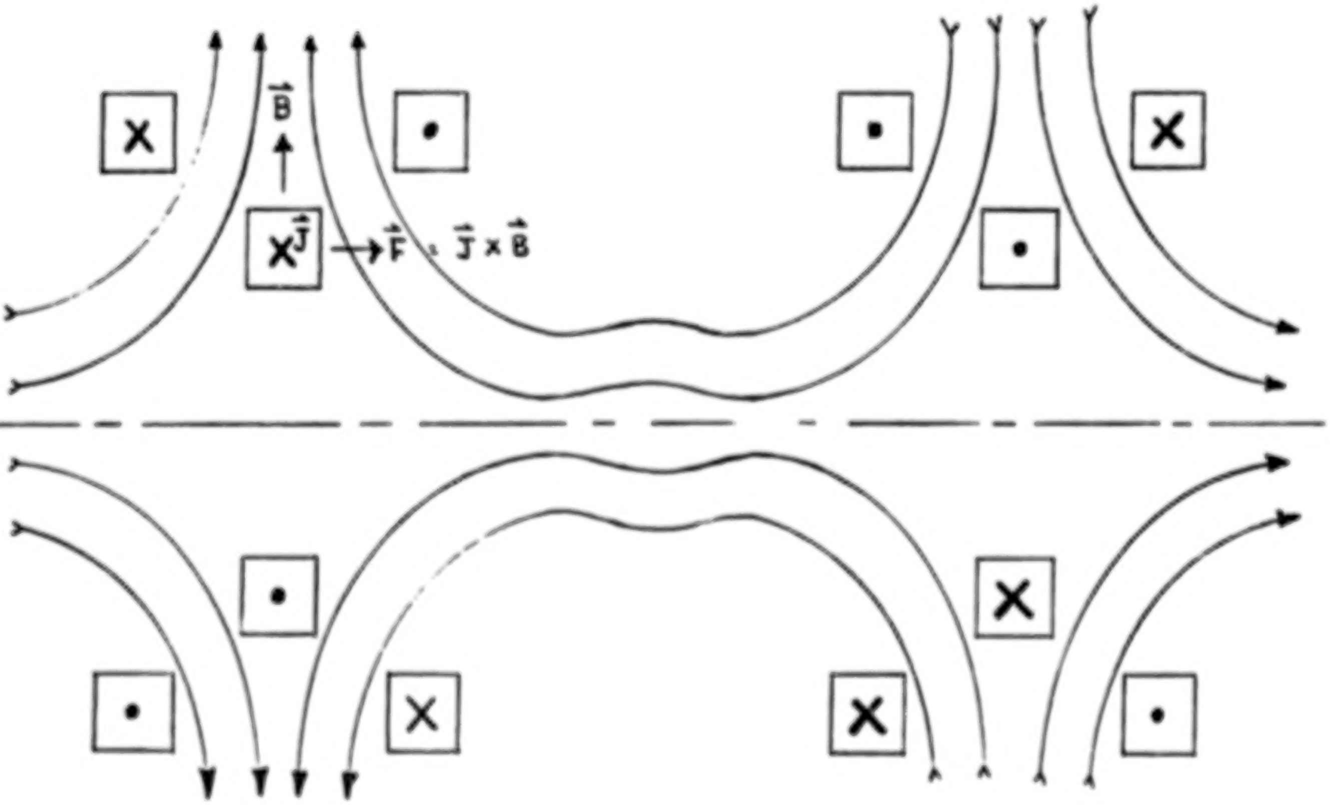
RELATIVE STRENGTHS OF INTERACTIVE FORCES



SPECIFIC IMPULSE	PROPELLANT
3×10^7	PHOTONS AND PARTICAL/ANTIPARTICLE ANNIHILATION
2900	He/He
2400	He/H
2129	H/H
974	CH/CH
834	N/N
577	H_2/O_3
567	O/O
523	H_2/O_2
446	H_2/O_2 (CENTAUR)

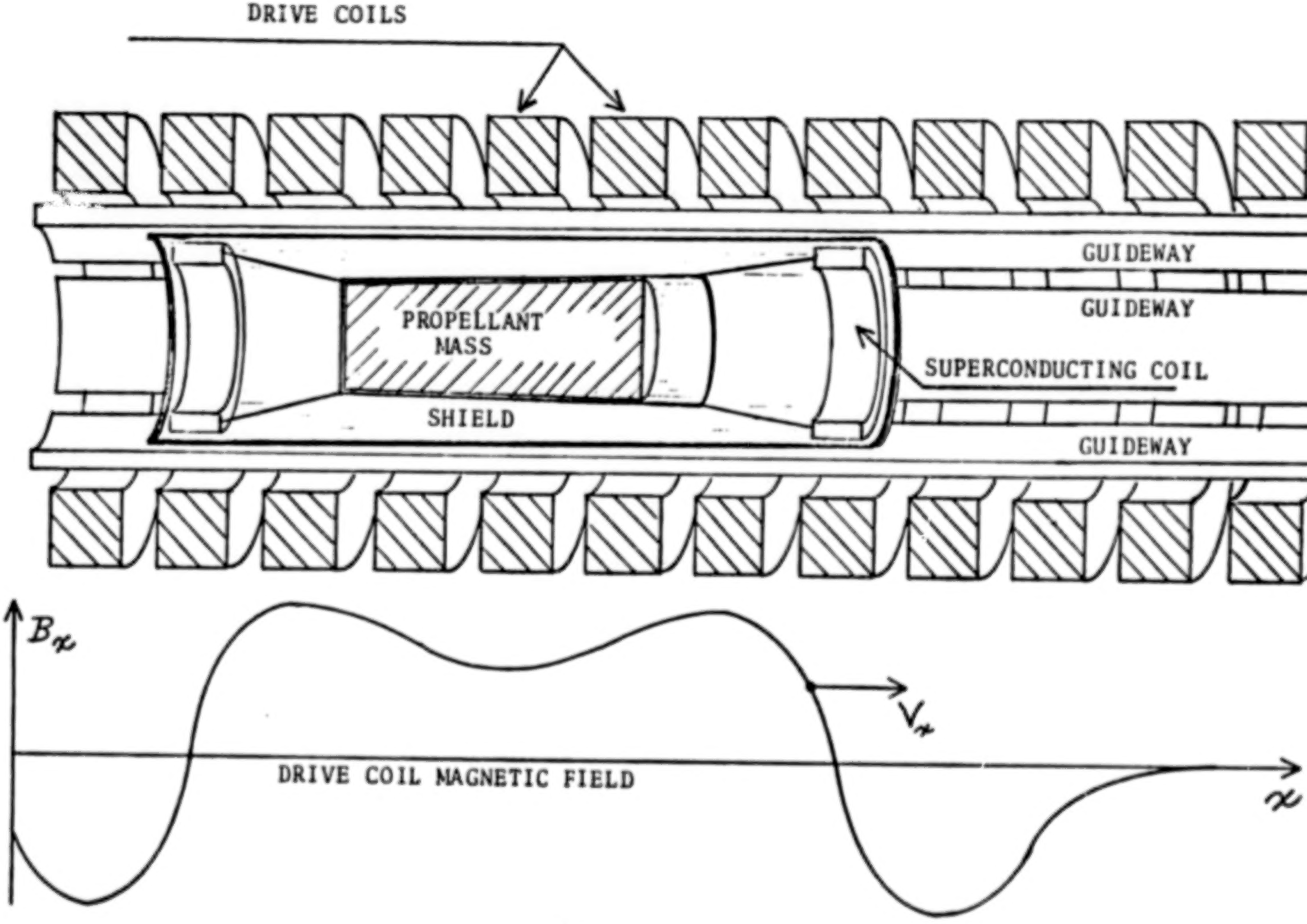
MASS DRIVER

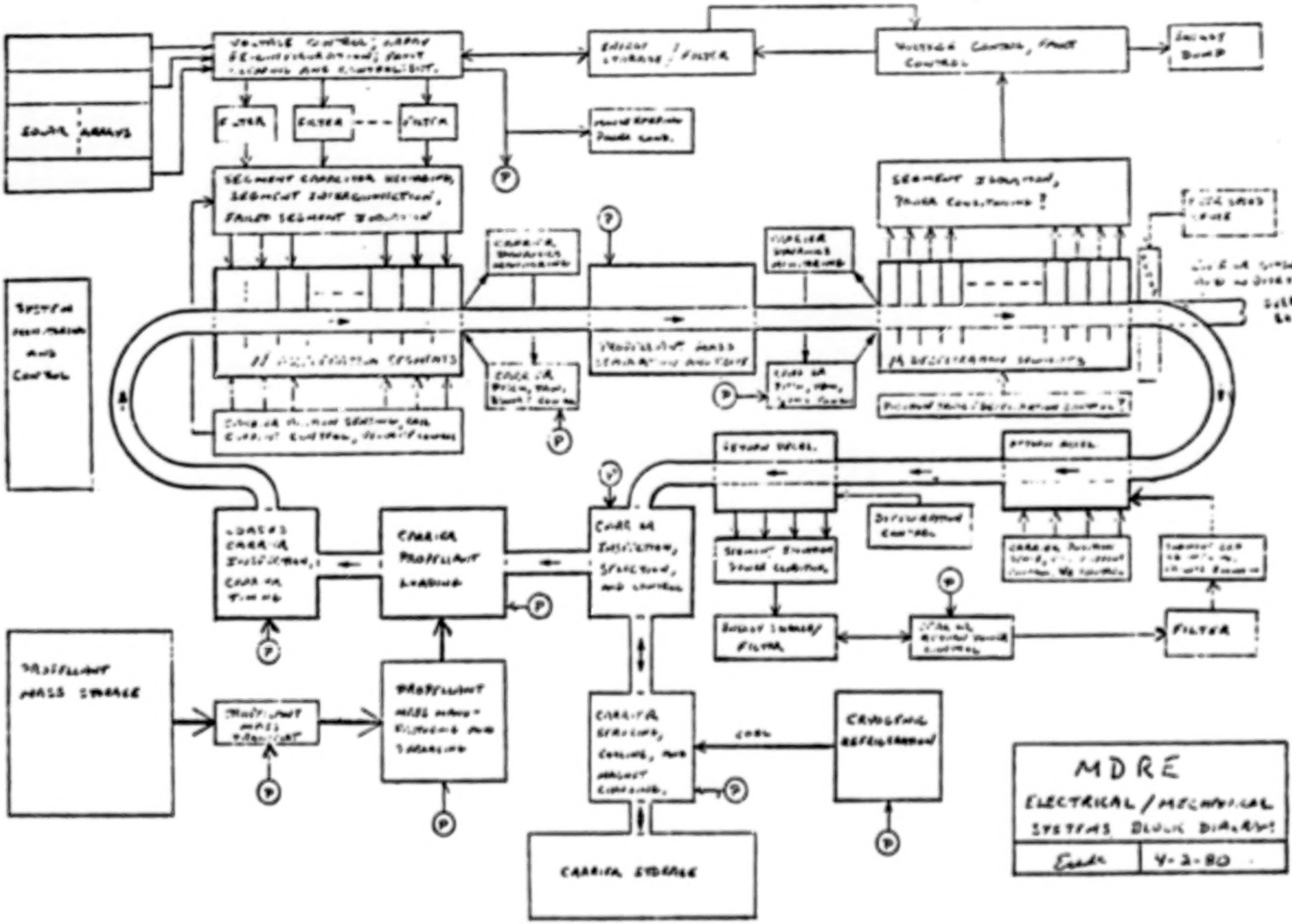
LINEAR SYNCHRONOUS MOTOR


$\vec{B}$
$\vec{J}$
$\vec{F} = \vec{J} \times \vec{B}$

MASS DRIVER

LINEAR SYNCHRONOUS MOTOR


DRIVE COILS
GUIDEWAY
GUIDEWAY
PROPELLANT MASS
SUPERCONDUCTING COIL
SHIELD
GUIDEWAY
B_x
V_x
DRIVE COIL MAGNETIC FIELD
x

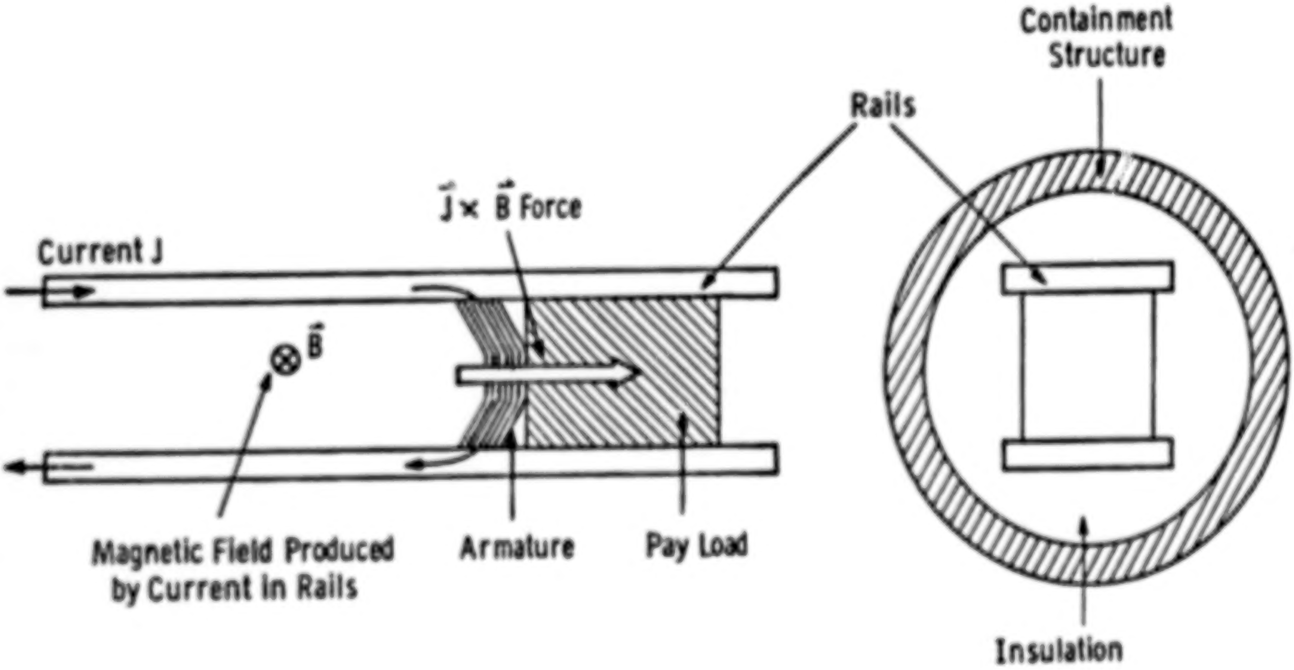


SOLAR ARRAYS
VOLTAGE CONTROL; ARRAY RECONFIGURATION; FAULT ISOLATION AND CONTROL UNIT.
ENERGY STORAGE / FILTER
VOLTAGE CONTROL, FAULT CONTROL
SHUNT DUMP
FILTER
FILTER
FILTER
HOUSEKEEPING POWER COND.
P
SEGMENT CAPACITOR RECHARGE, SEGMENT INTERCONNECTION, FAILED SEGMENT ISOLATION
SEGMENT ISOLATION, POWER CONDITIONING ?
FINE SPEED TRIM
SYSTEM MONITORING AND CONTROL
CARRIER DYNAMICS MONITORING
P
CARRIER DYNAMICS MONITORING
OVER SPEED EXIT
N ACCELERATION SEGMENTS
PROPELLANT MASS SEPARATION AND EXIT
N DECELERATION SEGMENTS
CARRIER PITCH, YAW, SHIFT CONTROL
CARRIER PITCH, YAW, SHIFT CONTROL
P
P
CARRIER POSITION SENSING, AND CURRENT CONTROL, SHIFT CONTROL
DECELERATION CONTROL ?
RETURN DECEL.
RETURN ACCEL.
DECELERATION CONTROL
SEGMENT ISOLATION POWER CONDITION.
CARRIER POSITION SENSING, CURRENT CONTROL
ENERGY STORAGE/ FILTER
FILTER
P
LOADED CARRIER INSPECTION, CARRIER TIMING
CARRIER PROPELLANT LOADING
CARRIER INSPECTION, SELECTION, AND CONTROL
P
P
PROPELLANT MASS STORAGE
PROPELLANT MASS TRANSPORT
P
PROPELLANT MASS MANUFACTURING AND SHAPING
P
CARRIER STORAGE, COOLING, AND MAGNET CHARGING.
COOL
CRYOGENIC REFRIGERATION
P
P
CARRIER STORAGE
MDRE
ELECTRICAL / MECHANICAL SYSTEMS BLOCK DIAGRAM
Esch
4-2-80

MASS DRIVER
LINEAR SYNCHRONOUS MOTOR
CHARACTERISTICS

INPUT POWER	$5 \times 10^7 - 7 \times 10^8$ WATTS
ASSUMED EFFICIENCY	70 - 95%
INSTANTANEOUS THRUST	10000N
THRUST TO POWER	200 mN/kW
FIRING FREQUENCY	1 - 10 Hz
SPECIFIC IMPULSE	1000 SEC
PROPELLANT ACCELERATION	1000 G's
PROPELLANT MASS	1 KG
LENGTH OF SYSTEM	8000 M
CALIBER (DRIVE COIL DIAM.)	20 CM

PRINCIPLE OF DIRECT CURRENT ELECTROMAGNETIC LAUNCHER


Containment Structure
Rails
$\vec{J} \times \vec{B}$ Force
Current J
$\otimes \vec{B}$
Magnetic Field Produced by Current in Rails
Armature
Pay Load
Insulation

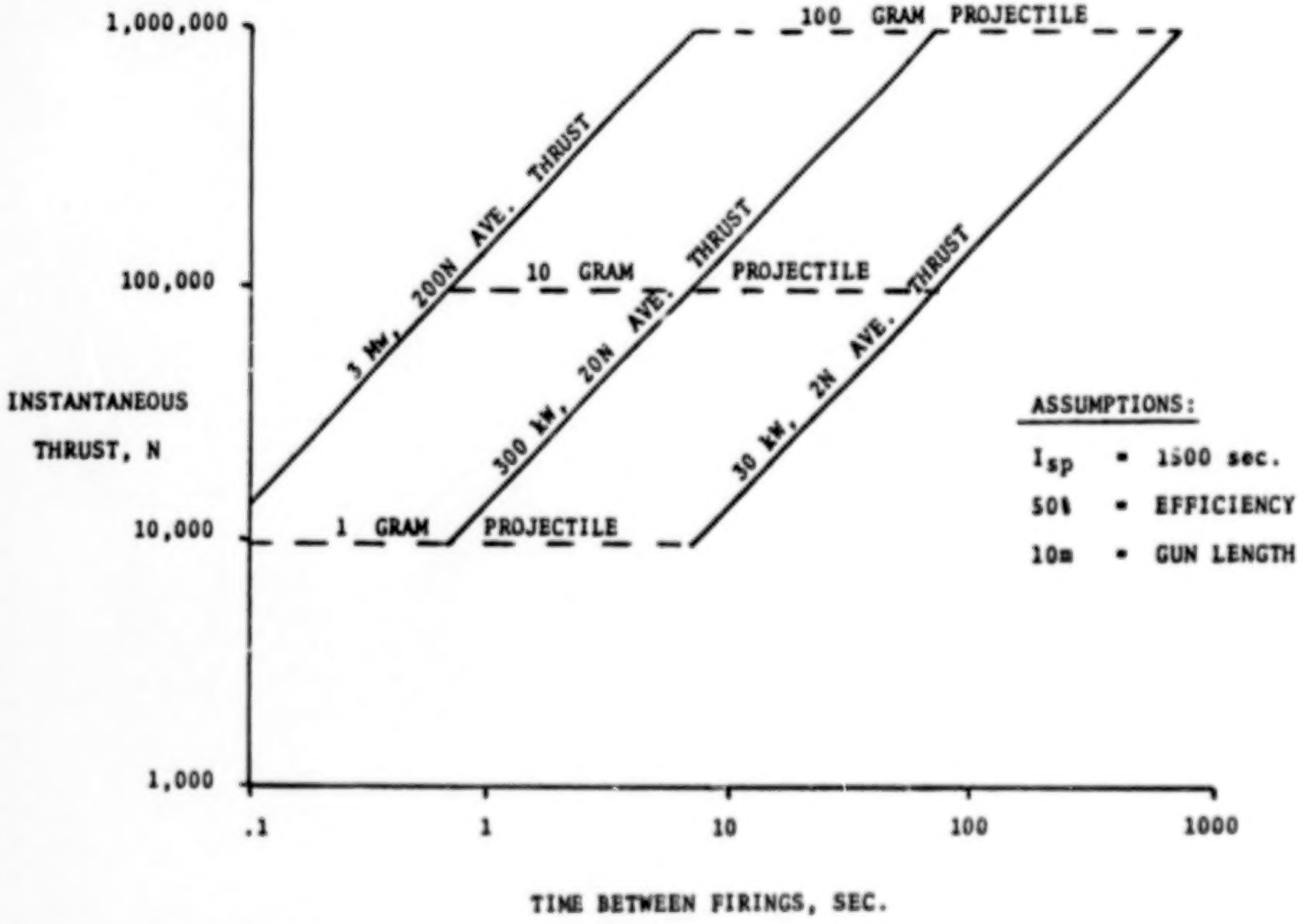
$$F = \tfrac{1}{2} L' I^2 = M \frac{dU}{dt}$$

RAIL GUN
CHARACTERISTICS

INPUT POWER	30 kW
ASSUMED EFFICIENCY	50 %
INSTANTANEOUS THRUST	96,000 N
THRUST TO POWER	102 MN/kW
FIRING FREQUENCY	.032 Hz
SPECIFIC IMPULSE	1000 SEC
PROPELLANT ACCELERATION	10^6 G's
PROPELLANT MASS	10 GRAMS
LENGTH OF SYSTEM	4.9 M

RAIL GUN

CHARACTERISTICS



MINUTES OF OPEN DISCUSSION

compiled by James J. Pelouch, Jr.
NASA Lewis Research Center

The following minutes have been interpreted from tape recordings and prepared statements. Individual identities are included when known; otherwise they are marked with a question mark (?).

Prepared Remarks by R. Carlisle:

Information presented confirms the existence of strong interactions between propulsion and LSS.

Future meetings of this type will be needed. They will have to emphasize progress towards a standard language for communications between propulsion and structures, and improved analytical tools for dealing with those interactions which appear to be very complex.

We will need to develop a list of those who maint in an interest in this area and make sure that they can participate in subsequent meetings.

This open discussion ought to focus on the following questions in order to surface future action items:

What are the interdisciplinary problems defined here?

What is the relative impact if different structural approaches are assumed?

Have the right questions and issues been raised?

We will now hear remarks from each moderator followed by audience response to these remarks.

Prepared Remarks by E. Gabris:

There was a general consensus on these broad issues:

1. A low-thrust chemical propulsion capability is required to transfer some types of large space structures from LEO to GEO. Based on structural considerations which limited the applied load to the 0.01 g region, the required thrust range is 500 to 3,000 lbf.

2. A propulsion system optimized for the low-thrust region could offer a 30 to 40 sec. improvement over the projected performance of a higher thrust system operated in a pump idle or tank-head idle mode. This difference corresponds to a payload data of approximately 2000 lbm to GEO. The "kitting" a high thrust engine for efficient low-thrust operation is a possibility that needs to be explored.

3. The optimum low-thrust level for a given system is sensitive to the assumptions regarding the structure, its packaging, and its use. These assumptions and their impact need to be clearly understood so effective comparative trades can be made between otherwise inconsistent study results.

4. No arguments were made for a distributed low-thrust chemical propulsion system.

5. A case for the use of electric propulsion for LEO to GEO transfer was not made, however one study suggested that it was not cost-effective for payloads less than 65K lbm.

6. It was suggested that a low-thrust system could be used in lieu of a high-thrust system with a moderate transfer time penalty (10-20 hours), or by multi-engine vehicles. The later could be advantageous if man-rating became a requirement. It was noted that a low-thrust propulsion system would not satisfy the requirements of planetary spacecraft as we know them today.

The following broad issues were identified:

1. Guidance is needed on the acceptable range of structural g loads to permit the propulsion design thrust range to be narrowed. It was suggested that the minimum gage structures which are the basis for many of the system studies will in practice be more rigid to satisfy control requirements, to permit analyses (the argument being that a structure which cannot be analyzed will never be built), to permit fabrication and assembly on the ground or to prevent damage when the structure is used or maintained in space. All such factors would permit a higher thrust system to be used.

2. Propulsion system designers need to determine the breaking point where the design of propulsion system components would change. Such knowledge would permit R&T activities to be directed to the most promising "low-thrust" range.

Finally a number of special questions were raised:

1. Are there mission strategies which would permit mission requirements to be satisfied in an optimum manner thus strengthening or weakening the case for low-thrust chemical or electric propulsion?

2. Will on-orbit requirements for station-keeping, attitude control, or intra-orbit mobility impact the selection and design of an inter-orbit propulsion system?

3. What is the trade between LEO assemble/deployment and checkout with low-g transfer to GEO, and GEO deployment following a high-g transfer? This trade should consider risk versus total cost.

4. Does a man's presence in GEO alter the above conclusions?

Audience Response to E. Gabris Remarks:

D. Byers - It is fundamentally important for us to understand what the mission level options and strategies are that make sense, and that this understanding will be of immense value to propulsion technology.

J. Hedgepeth - Many assumptions have been made with regard to electric propulsion that lead directly to hundred-day orbit transfer times. I know that it may (someday) be possible for a 5-day trip with electric propulsion. This requires electric propulsion systems whose specific power is about 2000 W/Kg, which is much higher than the systems currently being developed.

J. Pelouch -The cost and mass of space power plays as important part in the effectiveness of electric propulsion. It could be that space power technology belongs in the agenda for future meetings like this.

R. Finke - These meetings need more discussion to determine (propulsion) schemes for doing a total mission.

J. Pelouch - The idea of providing a single propulsion system to meet the combined requirements of orbit raising and on-orbit control is somewhat inconsistent with OAST's Space Systems Division Offices which now either emphasize orbit transfer propulsion (for vehicles) or auxiliary propulsion (as a spacecraft subsystem). Electric propulsion is the most logical candidate to meet the combined requirements because of the high specific impulse which is needed. A single system like this may be more cost effective than the sum of the costs of the separate systems which it would replace.

D. Byers - On-orbit propulsion (for LSS) requirements are more important than we think, and this fact may be getting lost here.

R. Carlisle - There is a fundamental need to use propulsion to control LSS on-orbit. It becomes an option to use this (on-orbit control) propulsion system also for orbit transfer, or to provide for a separate orbit transfer system.

J. Pelouch - Regarding deployment of LSS in LEO or deployment of LSS in GEO, the Shuttle will allow for functional checkout of LSS in LEO with humans prior to commitment of the LSS to some potentially irretrievable place in space. I think that the option to transfer LSS after deployment in LEO is preferred because of the expense and risk associated with deployment after orbit transfer to GEO.

J. Hedgepeth - This (LEO vs. GEO) issue certainly needs a tradeoff, relating to reliability of deployment and the cost of developing the deployment reliability into the LSS.

E. Gabris - It is clear to me that the GEO vs. LEO assumption has been challenged. We also need to know what humans can do in LEO to fix the spacecraft (if it fails to deploy and function).

R. Dergance - We also need to know "what LEO is". For instance, drag, delta-V, gravity gradient torque, etc. are grossly influenced by the altitude assumed for LEO, and these things, in turn, grossly influence the propulsion requirements. This is especially true for large spacecraft.

(?) - Addition of Orbiter OHM's kits also has a large effect on this.

R. Dergance - Also, it is important to recognize that the Shuttle performance is (now) not 60,000 pounds to LEO but instead maybe 16,000 pounds to LEO, and this also drives the propulsion requirements for LSS.

E. Gabris - Remember that we're talking of missions which are at least a decade away. I think that 60,000 pounds is a safe assumption.

Prepared Remarks by R. James (Audience Response Included):

(1) I wish to reconfirm the remark made by E. Gabris that deployment in LEO vs. deployment in GEO is an issue which is germane to this meeting. I must add that the spacecraft and mission concept in question will also influence the LEO vs. GEO deployment inssue.

(2) Concerning the utilization of humans to assist in deployment and repair of LSS, the LSST program is now supporting activities which are intended to quantify the man and machine role in such activities, and I expect that this information, as it evolves, will be of use in resolving some of the issues raised here.

(3) A general observation is that the propulsion/LSS interaction is not just steady state but also dynamic in nature, and that the problems which stem from dynamic considerations (such as POGO) may preclude the possibility of really "thin" structures.

(4) Another general observation is that there was no information provided on utilization of solar pressure for propulsion, perhaps for bringing hardware out of GEO. Rather than to counteract the solar pressure forces with propulsion, perhaps we should consider using these forces for propulsion.

Specific summary points and action items pertaining to the presentations given in my session are given below:

(1) Regarding Dergance's paper, he presented structure mass/size acceleration in relationships for 3 different structural concepts. For the hoop-column concept, he should compare notes with Harris Corp. and Langley. For the wrap-rib concept, he should compare notes with Lockheed and JPL. (Confirmed by all affected parties).

(2) Regarding Smith's paper, he should compare with results of a similar study by Hedgepeth. Also, several attendees raised questions on the aerodynamic force and thermal effects assumptions that Smith made.

(3) Regarding Walz's paper, his basic discussion pertained to frequency effects and their impact on platform design. What is the basis for the selection of a frequency to design to ?

J. Walz - We found that the on-orbit loads produced by the space environment to be so low as to preclude them from consideration in structural design so we instead considered a structural frequency that was 'sufficiently' high from a controls standpoint.

J. Hedgepeth - We had this experience too. If you put propulsive loads aside, the remaining loads are indeed so small as to be insignificant, and the design criteria then consists of producing high natural frequency. The exception to this may be the assembly loads.

R. Carlisle - I am concerned that control is not the only structural design criteria and that our studies are not recognizing the important ones (criteria). We may need to do some of these studies all over again.

(4) Regarding Hedgepeth's paper, the conclusions are very interesting and should be compared with Harris and Lockheed.

J. Pelouch - In particular, the frequency that a large antenna has to operate influences the surface shape accuracy which it must have which in turn influences the number and size of structural elements which it contains. The net effect is a reduced tolerance to propulsion loads compared to cases where antenna frequency is not a consideration.

(5) Regarding the Young/Vos papers, I feel that the integrated analysis capability (IAC) will give us the opportunity to provide a common working relationship between propulsion, structures, and controls.

J. Hedgepeth - Spacecraft dynamic design today is a major undertaking and we're not beginning to experience the size of spacecraft being talked about at this meeting. We are in a design environment where we cannot change a frozen design because of the complexity of the resulting dynamic interactions. We really need something like IAC.

J. Pelouch - I sense though, that IAC usefulness as a parametric tool is limited. (Confirmed by Young).

D. Byers - It concerns me that large sums of money are devoted to dynamic interactions when, with electric propulsion, the thrust is so low to eliminate this issue.

J. Pelouch - Indeed, the electric propulsion thrust may be high enough to make the dynamic problem persist, for all we know.

(6) Regarding Tolivar's paper, two important points are apparent. First, laboratory experiments are needed to verify models and to understand dynamic behavior and control techniques. Second, there is a need to extend the experimental environment into space (for large space system technology).

Prepared Remarks by Fred Teren

I have found that this meeting provided an excellent opportunity to exchange information. My observations are as follows:

(1) There are several important characteristics of propulsion that are applicable to transportation and control of LSS. For electric propulsion, the inherent high specific impulse which is in the 2000-10000 sec. range results in high payload fractions and introduces the possibility of reusing the propulsion system (on subsequent missions). The high electric propulsion system weight leads to low acceleration which leads to long trip times of 150-180 days. The low acceleration of electric propulsion, however, is very desirable for LSS. The technology for electric propulsion is pretty well characterized as evidenced by the status of the 8-cm and 30-cm systems. For chemical propulsion, the specific impulse is limited to below 500 sec. which means lower payload fractions (than with electric propulsion). This also makes reusability difficult and leads to concepts which consist of expendable chemical propulsion systems. Since the thrust is higher than with electric propulsion, the trip times with chemical propulsion systems are only several days while the acceleration is still low enough to preclude structural penalties to the large spacecraft. The technology for chemical propulsion for LSS is not as advanced as it is for electric propulsion.

(2) We will have to learn how to mechanize slow start in chemical propulsion systems if transient effects (LSS dynamic effects) are shown to be a problem.

(3) The combined modelling of the LSS and the propulsion system, from a dynamic standpoint, is required, but I perceive it to be a very difficult task to accomplish.

(4) I sense a need to further determine what the allowable g-level is for LSS. Much progress is apparent here, but I also see conflicting results. For instance, both MMC and GDC indicate an acceptable range of 0.05 to 0.1 g's, but Lewis and Langley indicate a benefit below 0.01 g's, perhaps even down to 0.001 g's. This difference appears to be due to different assumptions. We need to understand what these assumptions are and to reconcile the differences in them.

(5) I have observed that we need a way to realistically compare different propulsion technologies which are in various stages of development. Some of the propulsion systems discussed here exist as hardware and some exist as drawings on paper, and this introduces inequities in comparison. In addition, some propulsion concepts are best described as "far term" while others are "near term" which further aggrevates the comparative inequities.

Audience Response to F. Teren Remarks

R. Preston - In his presentation, Dave Byers noted that ion bombardment thrusters based on Lewis technology are the examples of electric propulsion maturity in this country. I must add that DOD has flown and is flying pulsed plasma electric propulsion systems which operate at a specific impulse of 2000 sec.

D. Byers - Indeed this is true. In addition, in the open literature, there is evidence of operational electric propulsion in Japan and the U.S.S.R.

R. Priem - The purpose of this meeting is to identify what new technology is needed in structures, not just in propulsion. The structural technology outlook will have an important bearing on the selection of propulsion technology. I think more information pertaining to structural technology would help this meeting.

F. Teren - I agree. This gets back to my statement about g-level inconsistencies and the fact that we in propulsion need to know what the g-level requirements are.

J. Hedgepeth - The difference in g-levels may be due to the peculiarity of individual payloads. The answer to what the g-level requirement is may indeed not exist or be unobtainable (because of the complexity of the problem). Perhaps what we need here is a "consensus of opinion" instead of analytical results. Speaking as a spacecraft designer, I would be very pleased to know that a particular g-level was available, so that I could proceed with spacecraft technology at that g-level.

(?) - In reference to Dr. Teren's comment on low-thrust chemical not being as technically advanced as electric, I wish to note that high chamber pressure, pressure fed chemical systems are well developed, even with hydrogen /oxygen.

(?) - Why is it concluded that low-thrust chemical propulsion ought to be expendable?

J. Pelouch - Recent economic studies at Lewis and elsewhere have shown that the cost of transporting the propellant to LEO which is necessary to return the chemical propulsion system from GEO to LEO (for reuse) exceeds the cost of the chemical propulsion system itself. This result is valid unless the launch cost is reduced by a factor of 10, or the specific impulse is increased to above 700 sec. Since neither of these events are credible, especially in the near term, it is concluded that chemical systems ought to be discarded in GEO.

D. Byers - I would like to add to the list of questions. Is spacecraft retrieval important or not? What about (spacecraft) disposal? What are the ground rules that we must associate with (LSS) missions? Propulsion people need to know these things.

Meeting adjourned at 3:00 PM on May 21, 1980 by R. Carlisle.

1. Report No. CP-2144 NASA CP 2144	2. Government Accession No.	3. Recipient's Catalog No.

4. Title and Subtitle LARGE SPACE SYSTEMS/LOW-THRUST PROPULSION TECHNOLOGY	5. Report Date July 1980
	6. Performing Organization Code
7. Author(s)	8. Performing Organization Report No. E-510
	10. Work Unit No.
9. Performing Organization Name and Address National Aeronautics and Space Administration Lewis Research Center Cleveland, Ohio 44135	11. Contract or Grant No.
	13. Type of Report and Period Covered Conference Publication
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546	14. Sponsoring Agency Code

15. Supplementary Notes

16. Abstract

A two-day meeting to discuss the interactions between space propulsion and large space systems (LSS) was held on May 20 and 21, 1980, at the NASA Lewis Research Center, Cleveland, Ohio. Twenty-two presentations were made by NASA Headquarters, NASA Centers, and Industry in three topical areas: Systems/Mission Analysis, LSS Static and Dynamic Characterization, and Propulsion System Characterization. These presentations provided an interdisciplinary exchange of information among propulsion, structures and materials, and controls, with emphasis on large spacecraft and missions. An open discussion of the issues took place at the end of the meeting. This publication contains the presentations and minutes of the open discussion.

17. Key Words (Suggested by Author(s)) Large space systems; Space platform systems; Structural analysis; Auxiliary propulsion; Electric propulsion; Low-thrust chemical propulsion; Space missions	18. Distribution Statement Unclassified - unlimited STAR Category 20

19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 354	22. Price* A16

* For sale by the National Technical Information Service, Springfield, Virginia 22161

350

*USGPO: 1980 — 657-145/405

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8-23-81